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# Advanced Oxygen-Hydrocarbon Rocket Engine Study

Contract NAS 8-33452
Bi-Monthly Progress Report 33452-M-4
June 1980

Prepared For:
National Aeronautics And Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

By:
C. J. O'Brien



Aerojet
Liquid Rocket
Company

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## FOREWORD

This is the fourth bi-monthly progress report submitted for the Advanced Oxygen - Hydrocarbon Rocket Engine Study per the requirements of Contract NAS 8-33452. The work is being performed by the Aerojet Liquid Rocket Company for the NASA-Marshall Space Flight Center. The contract was issued on 15 October 1979. The program inclusive dates for period of performance are 15 October 1979 through 15 February 1981. This report covers the period from 1 April 1980 to 31 May 1980.

The program consists of parametric analysis and design to provide a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure $LO_2$/Hydrocarbon engine systems.

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R. Salkeld - Vehicle Trajectory Performance Assessment

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## I. INTRODUCTION

In the decade of the 1980's and beyond, the nation's expanding space operations may require an improved surface-to-orbit transportation system using advanced booster vehicles which have increased performance and capability compared to the current space shuttle concept. The mixed-mode propulsion principle clearly indicates the potential performance advantages of using high density-impulse rocket propellants in such large $\Delta V$ applications. For this reason, hydrocarbon fuels exhibiting increased density relative to liquid hydrogen ($LH_2$), at the penalty of lower specific impulse, are being considered for the booster propulsion system of space shuttle improvements and derivatives as well as for single-stage-to-orbit and two-stage-to-orbit heavy-payload vehicles.

Preliminary identification and evaluation of promising liquid oxygen/hydrocarbon ($LO_2$/HC) rocket engine cycles is desirable to produce a consistent and reliable data base for vehicle optimization and design studies, to demonstrate the significance of propulsion system improvements, and to select the critical technology areas necessary to realize such advances.

It is the purpose of this study to generate a consistent engine system data base for defining advantages and disadvantages, system performance and operating limits, engine parametric data, and technology requirements for candidate high pressure $LO_2$/HC engine systems. The study will also synthesize optimum $LO_2$/HC engine power cycles and generate representative conceptual engine designs for a specified advanced surface-to-orbit transportation system.

To accomplish the program objectives, the study is composed of four major technical tasks and a reporting task. These tasks and summarized objectives are:

### A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

Formulate and assess families of high chamber pressure $LO_2$/HC engine cycles.

I. Introduction (cont.)

B. TASK II - ENGINE PARAMETRIC ANALYSIS

Generate performance, weight, and envelope parametric data for viable concepts based upon historical data and conceptual evaluations.

C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT (ENGINE SCREENING)

Conduct a preliminary comparison of selected engine cycles utilizing a simplified vehicle trajectory performance model.

D. TASK IV - BASELINE ENGINE SYSTEMS DEFINITION

Prepare preliminary designs of two baseline engine configurations. Conduct heat transfer, turbomachinery, combustion stability, structural, and controls analysis of the baseline engines and components. Conduct a parametric sensitivity analysis including the effects of turbine temperature and number of usable life cycles. Provide the appropriate data in a format suitable for use in vehicle application analyses.

E. TASK V - REPORTING

Provide informal bi-monthly technical and fiscal progress reports, hold program reviews at NASA/MSFC and prepare a final report.

II. TECHNICAL PROGRESS SUMMARY

The overall progress on the program is indicated in Figure 1.

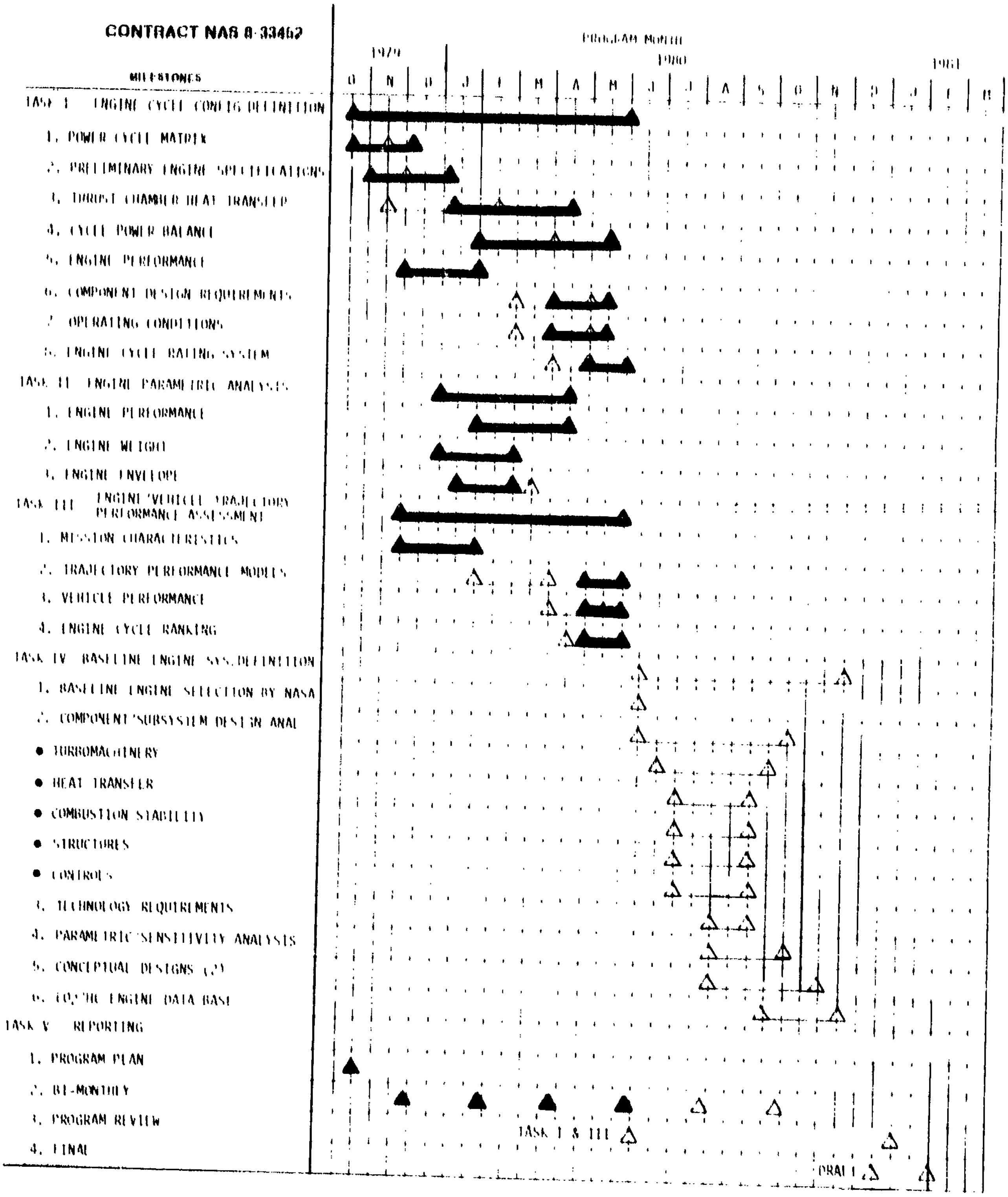



Figure 1. Major Milestone Schedule

A. TASK I - ENGINE CYCLE CONFIGURATION DEFINITION

1. Cycle Power Balance

Power balance data were generated for the eleven engine cycles given in Table I, utilizing the parametric heat transfer and performance data summarized in Bi-Monthly Progress Report 33452M-3. The cycles labelled A through I, as shown in Figure 2, are those specified by the contract. Cycles J and K were selected as the most promising additional candidates from the preliminary power cycle studies previously conducted and summarized in Bi-Monthly Progress Report 33452M-2.

Engine specification data based on the parametric performance data are given in Tables II and III. These data were used for the power balance evaluation of staged combustion cycle engines and the thrust chambers of open-loop (e.g., gas generator) cycle engines. In many cases, power balances were not achieved at the higher chamber pressures (Pc = 3000 to 5000) due to cooling limitations. The power cycle data are summarized in Table IV, and are interpreted in the following paragraphs.

a. Cycle A

The schematic for engine cycle A is given in Figure 3. The results from the power balance analysis are summarized in Figures 4 and 5. Calculations were initially made from coolant pressure drop data obtained by assuming no carbon deposit on the hot gas-side chamber wall, and by assuming a maximum local coolant-side wall temperature of 550°F, the coking (or decomposition) temperature for MIL Spec RP-1. With no carbon deposition, the RP-1 coolant pressure drop at a chamber pressure of 1000 psia and an engine thrust level of 200,000 lbf is 1600 psia. Since it would not be possible to achieve an engine power balance with RP-1 at much higher chamber pressures (see Figure 4), deoxygenated and purified RP-1 was utilized in most of the calculations. The purified fuel is similar to JP-5 which has a decomposition temperature of 800°F.

TABLE I

ENGINE CYCLE DESCRIPTION

| DESIGNATION | PROPELLANTS | COOLANT | CYCLE TYPE |
|---|---|---|---|
| A | $LO_2$/RP-1 | RP-1 | RP-1 fuel-rich (FR) gas generator (GG) |
| B | $LO_2$/RP-1 | $LO_2$ | RP-1 FR GG |
| C | $LO_2$/$LCH_4$ | $LCH_4$ | $LCH_4$ FR GG |
| D | $LO_2$/RP-1 | RP-1 | RP-1 FR preburner (PB) staged combustion (SC) |
| E | $LO_2$/RP-1 | $LO_2$ | RP-1 FR PB SC |
| F | $LO_2$/RP-1 | RP-1 | $LO_2$ oxidizer-rich (OR) PB SC |
| G | $LO_2$/RP-1 | $LO_2$ | $LO_2$ OR PB SC |
| H | $LO_2$/$LCH_4$ | $LCH_4$ | $LCH_4$ FR PB SC |
| I | $LO_2$/$LCH_4$ | $LCH_4$ | FR and OR PB SC |
| J | $LO_2$/RP-1 | $LH_2$ | $LH_2$ FR GG |
| K | $LO_2$/$LCH_4$ | $LH_2$ and $LCH_4$ | $LH_2$ FR GG and 2 OR PB SC Dual Throat |

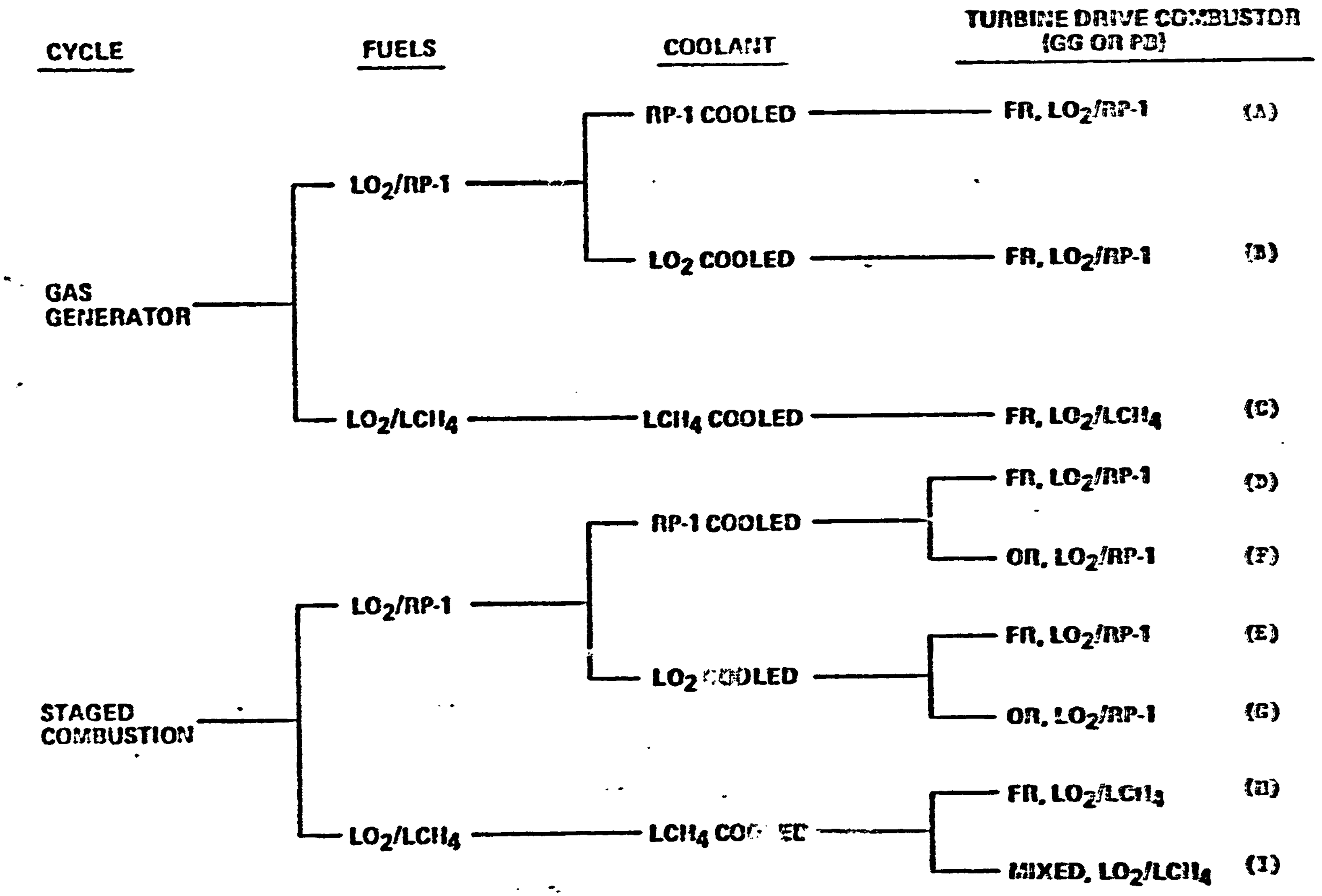



Figure 2. Candidate Cycles for Advanced $LO_2$/HC Engines

# TABLE II

## $LO_2$/RP-1 THRUST CHAMBER ASSEMBLY SPECIFICATION DATA

| PARAMETER | | | | | |
|---|---|---|---|---|---|
| Chamber Pressure, psia | 5000 | 4000 | 3000 | 2000 | 1000 |
| Thrust, sl, lbf | 600,000 | 600,000 | 600,000 | 600,000 | 600,000 |
| Thrust, vac, lbf | 662,617 | 666,433 | 671,605 | 679,365 | 695,618 |
| Mixture Ratio | 2.8 | 2.8 | 2.8 | 2.8 | 2.8 |
| Area Ratio | 61.7 | 51.9 | 41.3 | 29.8 | 17.2 |
| ODE Is, sl, sec | 334.4 | 330.7 | 325.3 | 313.6 | 293.5 |
| ODE Is, vac, sec | 368.1 | 365.9 | 362.5 | 353.7 | 339.5 |
| Is Efficiency, %(V) | 96.7 | 96.6 | 96.4 | 96.8 | 96.7 |
| Deliv. Is, sl, sec | 322.3 | 318.2 | 312.2 | 302.4 | 283.0 |
| Deliv. Is, vac, sec | 355.9 | 353.4 | 349.4 | 342.4 | 328.1 |
| Total Flowrate, lb/s | 1861.62 | 1885.61 | 1921.84 | 1984.13 | 2120.14 |
| $LO_2$ Flowrate, lb/s | 1371.72 | 1389.39 | 1416.10 | 1461.99 | 1562.21 |
| Fuel Flowrate, lb/s | 489.90 | 496.21 | 505.75 | 522.14 | 557.93 |
| c*, ft/s | 5958 | 5945 | 5922 | 5897 | 5850 |
| Throat Area, $in^2$ | 69.10 | 86.92 | 118.05 | 181.8 | 385.5 |
| Throat Diam., in. | 9.38 | 10.52 | 12.26 | 15.21 | 22.15 |
| Exit Area, $in^2$ | 4259 | 4520 | 4876 | 5418 | 6631 |
| Exit Diam., in | 73.6 | 75.9 | 78.8 | 83.1 | 91.9 |
| Exit Pressure, psia | 6.0 | 6.0 | 6.0 | 6.0 | 6.0 |

TABLE III

$LO_2/LCH_4$ THRUST CHAMBER ASSEMBLY SPECIFICATION DATA

| PARAMETER | | | | | |
|---|---|---|---|---|---|
| Chamber Pressure, psia | 5000 | 4000 | 3000 | 2000 | 1000 |
| Thrust, sl, lbf | 600,000 | 600,000 | 600,000 | 600,000 | 600,000 |
| Thrust, vac, lbf | 665,090 | 668,416 | 673,553 | 682,193 | 700,698 |
| Mixture Ratio | 3.5 | 3.5 | 3.5 | 3.5 | 3.5 |
| Area Ratio | 63.9 | 53.2 | 42.2 | 30.5 | 17.6 |
| ODE Is, sl, sec | 343.1 | 338.3 | 332.4 | 321.5 | 301.2 |
| ODE Is, vac, sec | 378.8 | 375.4 | 371.6 | 363.8 | 349.8 |
| Is Efficiency, %(V) | 96.4 | 96.6 | 96.5 | 96.7 | 96.5 |
| Deliv. Is, sl, sec | 329.6 | 325.4 | 319.4 | 309.4 | 289.0 |
| Deliv. Is, vac, sec | 365.3 | 362.5 | 358.5 | 351.7 | 337.5 |
| Total Flowrate, lb/s | 1820.39 | 1843.88 | 1878.52 | 1939.24 | 2076.12 |
| $LO_2$ Flowrate, lb/s | 1415.86 | 1434.13 | 1461.07 | 1508.30 | 1614.76 |
| Fuel Flowrate, lb/s | 404.53 | 409.75 | 417.45 | 430.94 | 461.36 |
| c*, ft/s | 6119 | 6107 | 6088 | 6063 | 6017 |
| Throat Area, $in^2$ | 69.2 | 87.5 | 118.5 | 182.7 | 388.3 |
| Throat Diam., in | 9.39 | 10.56 | 12.28 | 15.25 | 22.24 |
| Exit Area, $in^2$ | 4425 | 4655 | 5000 | 5573 | 6833 |
| Exit Diam., in | 75.1 | 77.0 | 79.8 | 84.2 | 93.3 |
| Exit Pressure, psia | 6.0 | 6.0 | 6.0 | 6.0 | 6.0 |

TABLE IV (1 of 4)

POWER CYCLE MATRIX FOR OXYGEN - HYDROCARBON ROCKET ENGINES

SEA LEVEL THRUST - 600,000 LBF

| Engine Cycle | Fuel | Coolant | Turbine Drive | Turbine Gas Temp. (°F) | Chamber Pressure (psia) | Pump Dischg. Pressure (psia) | S.L. Specific Impulse (sec) | Vac. Specific Impulse (sec) | Comments |
|---|---|---|---|---|---|---|---|---|---|
| (a) Gas Generator | RP-1 | RP-1 | RP-RICH | 1400 | 1000 | 1441 | 279.7 | 324.8 | No carbon deposit on chamber wall. Purified RP-1 with 800°F coking temperature. |
| | | | | | 2000 | 3629 | 294.6 | 334.5 | |
| | | | | | 2500 | 5095 | 297.4 | 335.8 | |
| | | | | 1800 | | | 301.3 | 339.9 | |
| | | | | 2500 | | | 305.8 | 344.5 | |
| | | | | 1400 | 2000 | 2671 | 295.5 | 335.4 | Carbon deposit on chamber wall. Purified RP-1 with 800°F coking temperature. |
| | | | | | 2500 | 3762 | 298.6 | 337.0 | |
| | | | | | 3000 | 5406 | 300.0 | 337.1 | |
| | | | | 1800 | 2500 | 3762 | 302.0 | 340.7 | |
| | | | | | 3000 | 5406 | 304.7 | 342.0 | |
| | | | | 2500 | | | 309.4 | 347.0 | |
| | | | | 1400 | 2000 | 3717 | 294.5 | 334.4 | Carbon deposit. MIL.SPEC. RP-1 with 550°F coking temperature. |
| | | | | 1800 | | | 297.4 | 337.5 | |
| (B) Gas Generator | RP-1 | $LO_2$ | RP-RICH | 1400 | 1000 | 1320 | 279.7 | 324.9 | No carbon deposit on chamber wall. Optimum performance at PC of 3000 to 3500. |
| | | | | | 2000 | 2389 | 294.9 | 334.8 | |
| | | | | | 3000 | 4774 | 299.7 | 336.7 | |
| | | | | | 4000 | 7797 | 298.9 | 333.8 | |
| | | | | 1800 | 3000 | 4774 | 304.5 | 341.8 | |
| | | | | | 4000 | 7797 | 306.4 | 341.7 | |
| | | | | 1400 | 3000 | 4269 | 300.6 | 337.6 | Carbon deposit on chamber wall. |
| | | | | | 4000 | 6700 | 300.8 | 335.7 | |
| | | | | | 5000 | 11147 | 296.1 | 329.3 | |
| | | | | 1800 | 3000 | 4269 | 305.1 | 342.4 | |
| | | | | | 4000 | 6700 | 307.6 | 342.8 | |
| | | | | 2500 | 3000 | 4269 | 309.5 | 347.1 | |
| | | | | | 4000 | 6700 | 314.1 | 349.8 | |

TABLE IV (2 of 4)

POWER CYCLE MATRIX FOR OXYGEN - HYDROCARBON ROCKET ENGINES

SEA LEVEL THRUST - 600,000 LBF

| Engine Cycle | Fuel | Coolant | Turbine Drive | Turbine Gas Temp. (°F) | Chamber Pressure (psia) | Pump Dischg. Pressure (psia) | S.L. Specific Impulse (sec) | Vac. Specific Impulse (sec) | Comments |
|---|---|---|---|---|---|---|---|---|---|
| (C)Gas Generator | $LCH_4$ | $LCH_4$ | $CH_4$-RICH | 1400 | 1000 | 1263 | 287.5 | 336.2 | No carbon deposit. Optimum performance at Pc = 4000. |
| | | | | | 2000 | 2700 | 305.9 | 342.6 | |
| | | | | | 3000 | 4447 | 313.5 | 353.1 | |
| | | | | | 4000 | 6921 | 316.5 | 354.1 | |
| | | | | | 5000 | 10961 | 316.6 | 352.9 | |
| | | | | 1800 | 3000 | 4447 | 316.1 | 355.7 | No carbon deposit. Higher turbine temperature. Shifts optimum performance to higher Pc. |
| | | | | | 4000 | 6921 | 320.3 | 358.1 | |
| | | | | | 5000 | 10961 | 322.0 | 358.6 | |
| | | | | 2500 | 3000 | 4447 | 316.7 | 356.2 | |
| | | | | | 4000 | 6921 | 321.5 | 359.1 | |
| | | | | | 5000 | 10961 | 324.1 | 360.5 | |
| | | | | 1400 | 4000 | 7030 | 315.8 | 353.5 | S.L. Thrust = 200,000 lbf |
| | | | | | | 7662 | 316.5 | 354.1 | S.L. Thrust = 1,500,000 lbf |
| (D)Staged Combustion | RP-1 | RP-1 | RP-RICH | 1400 | 1000 | 1840 | 283.0 | 328.1 | No carbon deposit. Purified RP-1 with 800°F coking temperature. |
| | | | | | 2000 | 5303 | 302.4 | 342.4 | |
| | | | | | 2500 | 8188 | 307.8 | 346.4 | |
| | | | | 1800 | 2000 | 4610 | 302.4 | 342.4 | |
| | | | | | 2500 | 6645 | 307.8 | 346.4 | |
| | | | | 1400 | 2000 | 4163 | 302.4 | 342.4 | Carbon deposit on chamber wall. Purified RP-1 with 800°F coking temperature. |
| | | | | | 2500 | 6397 | 307.8 | 346.4 | |
| | | | | | 3000 | 10284 | 312.2 | 349.4 | |
| | | | | 1800 | 2000 | 3570 | 302.4 | 342.4 | |
| | | | | | 2500 | 5170 | 307.8 | 346.4 | |
| | | | | | 3000 | 7572 | 312.2 | 349.4 | |
| | | | | 2500 | 3000 | 6481 | 312.2 | 349.4 | |
| | | | | 1400 | 2000 | 5407 | 302.4 | 342.4 | Carbon deposit. MIL. Spec. RP-1 with 550°F coking temperature |
| | | | | 1800 | 2000 | 4698 | 302.4 | 342.4 | |

POWER CYCLE MATRIX FOR OXYGEN - HYDROCARBON ROCKET ENGINES

SEA LEVEL THRUST - 600,000 LBF

| Engine Cycle | Fuel | Coolant | Turbine Drive | Turbine Gas Temp. (°F) | Chamber Pressure (psia) | Pump Dischg. Pressure (psia) | S.L. Specific Impulse (sec) | Vac. Specific Impulse (sec) | Comments |
|---|---|---|---|---|---|---|---|---|---|
| (E) Staged Combustion | PP-I | $LO_2$ | RP-RICH | 1400 | 1000 | 1589 | 283.0 | 328.1 | No carbon deposit. Higher turbine temperature makes power balance possible at Pc = 4000. |
| | | | | | 2000 | 3966 | 302.4 | 342.4 | |
| | | | | | 3000 | 8524 | 312.2 | 349.4 | |
| | | | | 1800 | 4000 | 9517 | 318.2 | 353.4 | |
| | | | | 2500 | | 6570 | | | Carbon deposit on chamber wall. |
| | | | | 1800 | | 8995 | | | |
| | | | | 2500 | | 6485 | | | |
| (F) Staged Combustion | PP-I | PP-I | LOX-RICH | 1200 | 1000 | 1523 | 283.0 | 328.1 | Power balance limited to Pc = 2500 without carbon deposit and high temperature turb. |
| | | | | | 2000 | 3629 | 302.4 | 342.4 | |
| | | | | | 2500 | 5095 | 307.8 | 346.4 | |
| (G) Staged Combustion | PP-I | $LO_2$ | LOX-RICH | 1200 | 1000 | 1631 | 283.0 | 328.1 | No carbon deposit. |
| | | | | | 2000 | 3885 | 302.4 | 342.4 | |
| | | | | | 3000 | 7248 | 312.2 | 349.4 | |
| | | | | | 4000 | 14936 | 318.2 | 353.4 | |
| | | | | | | 12527 | | | Carbon deposit. |
| | | | | 2140 | | 10826 | | | No carbon deposit. |
| (H) Staged Combustion | $LCH_4$ | $LCH_4$ | $CH_4$-RICH | 1400 | 1000 | 1607 | 289.0 | 337.5 | No carbon deposit. Higher turbine temperature. Makes power balance possible at Pc greater than 3000. |
| | | | | | 2000 | 3945 | 309.4 | 351.7 | |
| | | | | | 3000 | 7896 | 319.4 | 358.5 | |
| | | | | 2500 | | 5517 | | | |
| | | | | 1800 | 4000 | 10794 | 325.4 | 362.5 | |
| | | | | 2500 | | 8728 | | | |

## TABLE IV (4 of 4)

POWER CYCLE MATRIX FOR OXYGEN - HYDROCARBON ROCKET ENGINES

SEA LEVEL THRUST - 600,000 LBF

| Engine Cycle | Fuel | Coolant | Turbine Drive | Turbine Gas Temp. (°F) | Chamber Pressure (psia) | Pump Dischg. Pressure (psia) | S.L. Specific Impulse (sec) | Vac. Specific Impulse (sec) | Comments |
|---|---|---|---|---|---|---|---|---|---|
| (I) Staged Combustion | $LCH_4$ | $LCH_4$ | $CH_4$-RICH | 1400 | | | | | No carbon deposit. Higher temperature turbine decreases fuel pump discharge pressure but MR shift increase LOX pump dischg. pressure. |
| | | | LOX-RICH | 1200 | 1000 | 1525 | 289.0 | 337.5 | |
| | | | | | 2000 | 3481 | 309.4 | 351.7 | |
| | | | | | 3000 | 6253 | 319.4 | 358.5 | |
| | | | | | 4000 | 10959 | 325.4 | 362.5 | |
| | | | | 1800/1200 | | 9146 | | | |
| | | | | 2500/1200 | | 9704 | | | |
| (J) Gas Generator | RP-1 | $LH_2$ | $LH_2$-RICH | 1380 | 1000 | 1325 | 284.1 | 329.3 | No carbon deposit. Use of $LH_2$ provides easy power balance. |
| | | | | | 2000 | 2566 | 303.3 | 343.4 | |
| | | | | | 3000 | 3826 | 312.9 | 350.2 | |
| | | | | | 4000 | 5193 | 318.6 | 354.0 | |
| | | | | | 5000 | 6740 | [illegible] | 356.4 | |
| (K) Dual Throat GG/SC | $LCH_4$ | $LH_2$<br>$LCH_4$ | $LH_2$-RICH<br>LOX-RICH | 1380<br>1200 | 2800/4000(I)<br>4000 (II) | 7429 | 319.0 (I)<br>-- | 357.9 (I)<br>380.2 (II) | Variable Area Ratio provides high Mode II performance. |

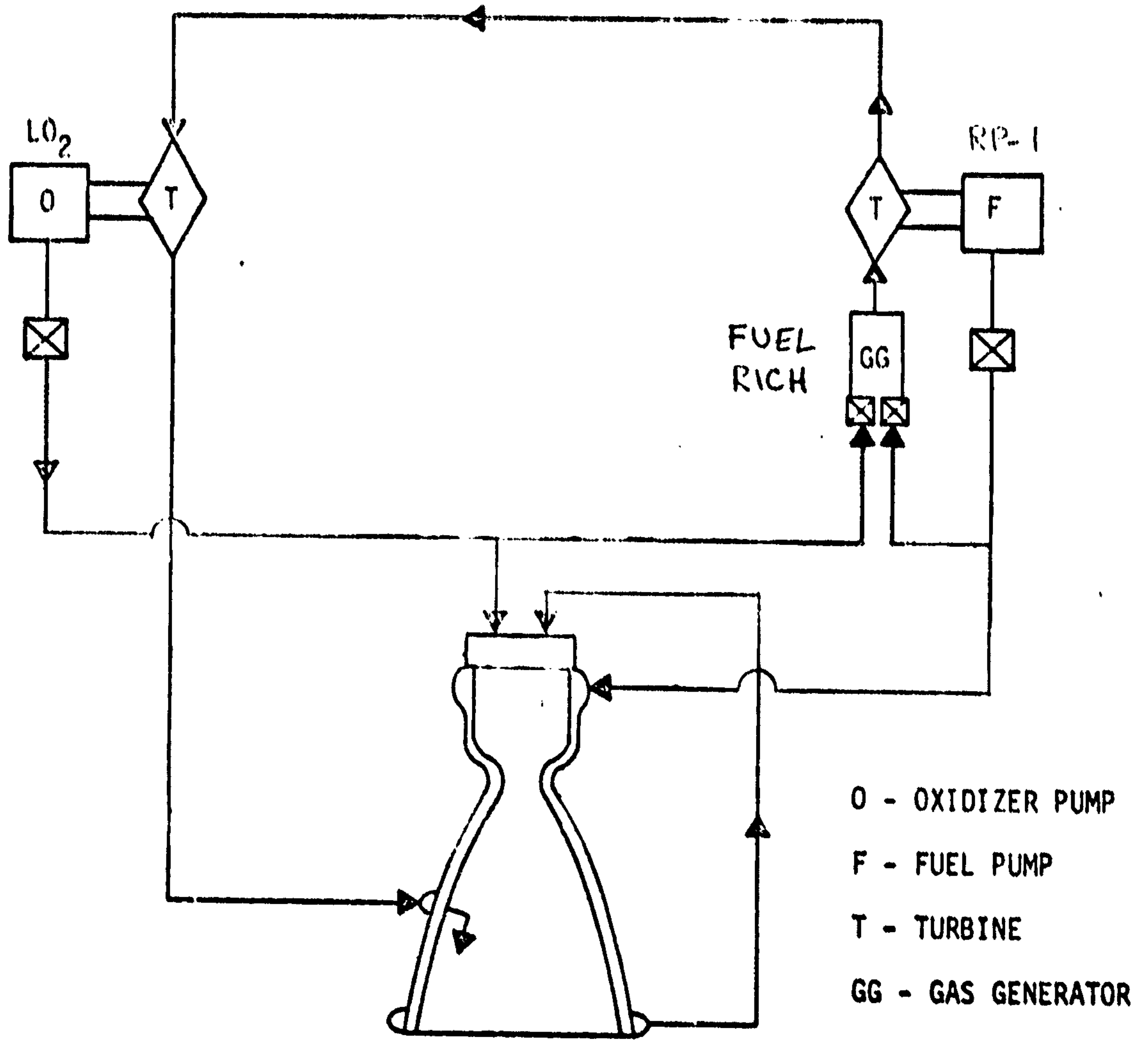



Figure 3. RP-1 Fuel-Rich Gas Generator Cycle (A) RP-1 Cooled

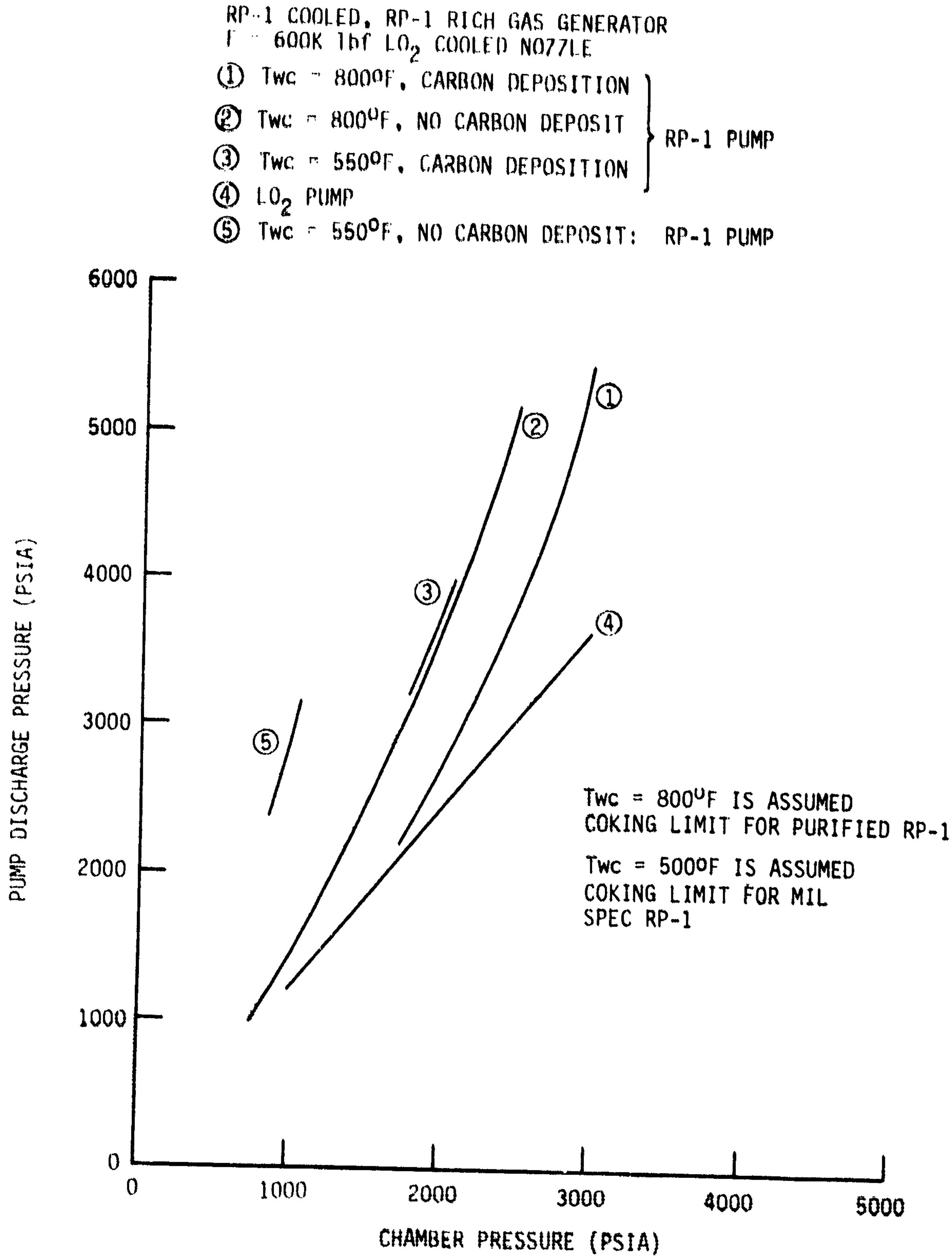



Figure 4. $LO_2$/RP-1 Engine Cycle A Pump Discharge Pressure Requirements

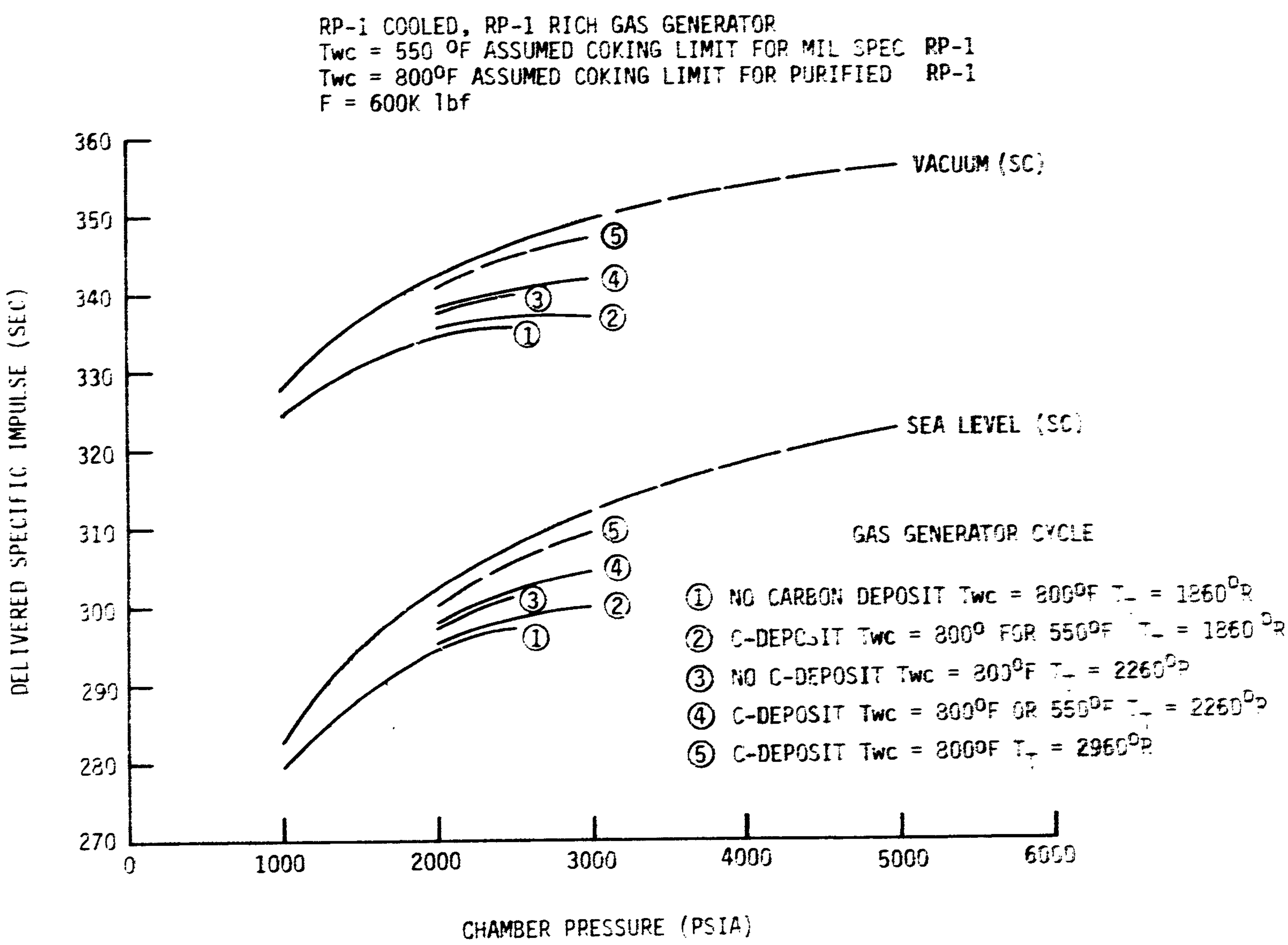



Figure 5. $LO_2$/RP-1 Engine Cycle A Performance

The conclusions revealed by Figure 4 are as follows: (1) MIL Spec RP-1 cooled engines (curve 5) are limited to a chamber pressure slightly above 1000 psia when there is no carbon deposit on the wall; (2) MIL Spec RP-1 cooled engines can achieve a chamber pressure in excess of 2000 psia (curve 3) if a unif..m carbon deposit is maintained on the chamber wall; (3) a purified RP-1 cooled engine (curve 2) is limited to a chamber pressure slightly in excess of 2500 psia without a carbon deposit; and (4) purified RP-1 cooled engines can achieve a chamber pressure (curve 1) in excess of 3000 psia if a carbon deposit is on the chamber wall.

The performance for the various cycle A engines is depicted in Figure 5, compared with a staged combustion cycle engine (see Table II). The conclusions to be reached in examining Figure 5 are: (1) a carbon deposit slightly influences (increases) the performance of the $LO_2$/RP-1 gas generator cycle engine (due to a reduction in gas generator flowrate); (2) the turbine inlet temperature has a large effect on gas generator engine performance (due to the variation in gas generator flowrate).

b. Cycle B

Cycle B differs from Cycle A in that $LO_2$ is used as the coolant. The schematic is given in Figure 6. The power balance results are summarized in Figures 7 and 8.

Figure 7 shows the significant benefit of using $LO_2$, rather than RP-1, as the coolant. If the 1980 state-of-the-art of rocket engine turbopumps is assumed to be 8,000 psia pump discharge pressure, then $LO_2$ is capable of cooling a $LO_2$/RP-1 engine with a chamber pressure of 4,000 psia (curve 2). When a carbon deposit is assumed, $LO_2$ is capable of cooling an engine with a chamber pressure of about 4,400 psia (curve 1).

Similar trends in performance are shown in Figure 8, as were seen in Figure 5. A carbon deposit provides a small increase in performance (about 1 second), and an increased turbine inlet temperature

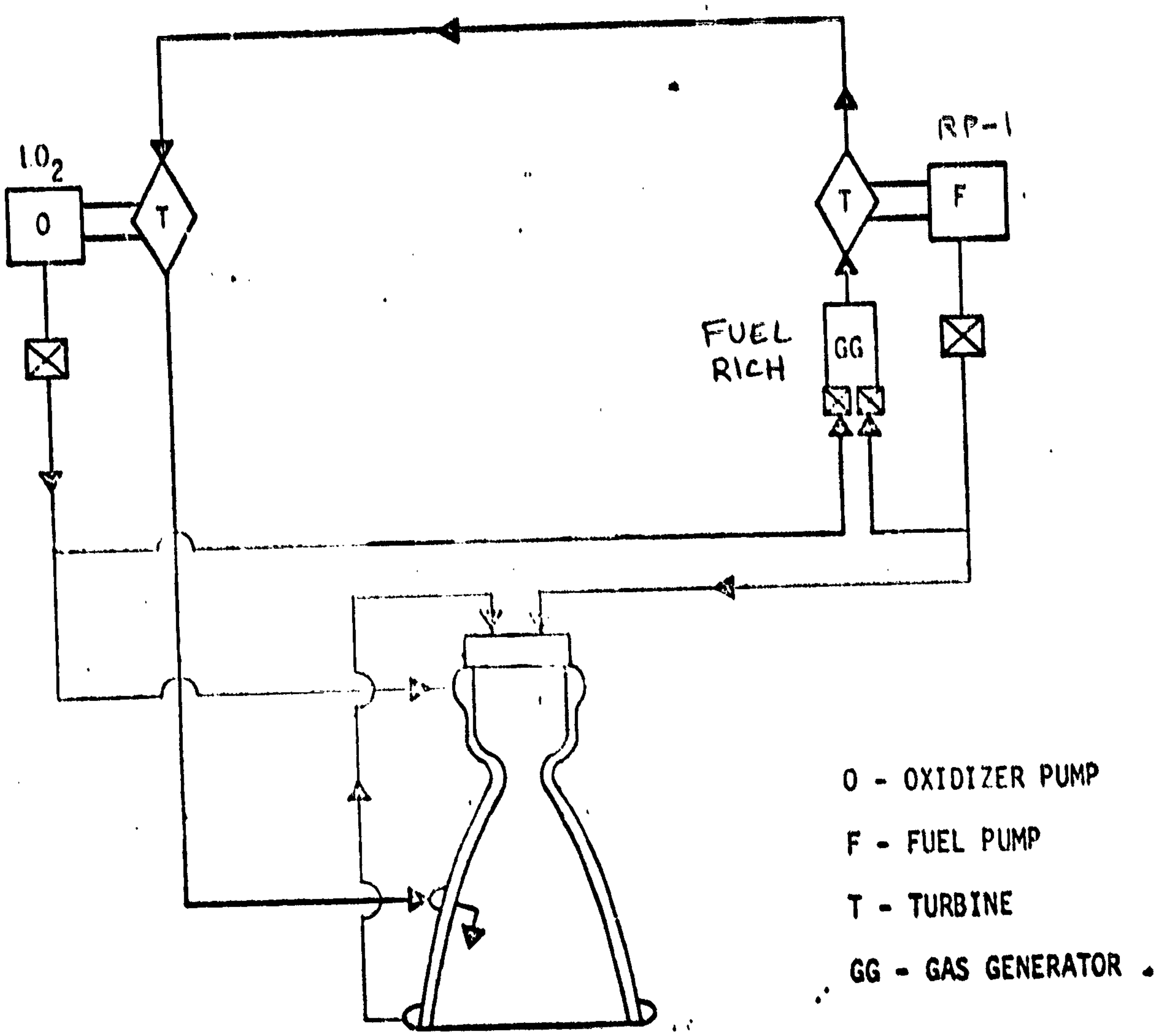



Figure 6. RP-1 Fuel-Rich Gas Generator Cycle (B) $LO_2$ Cooled

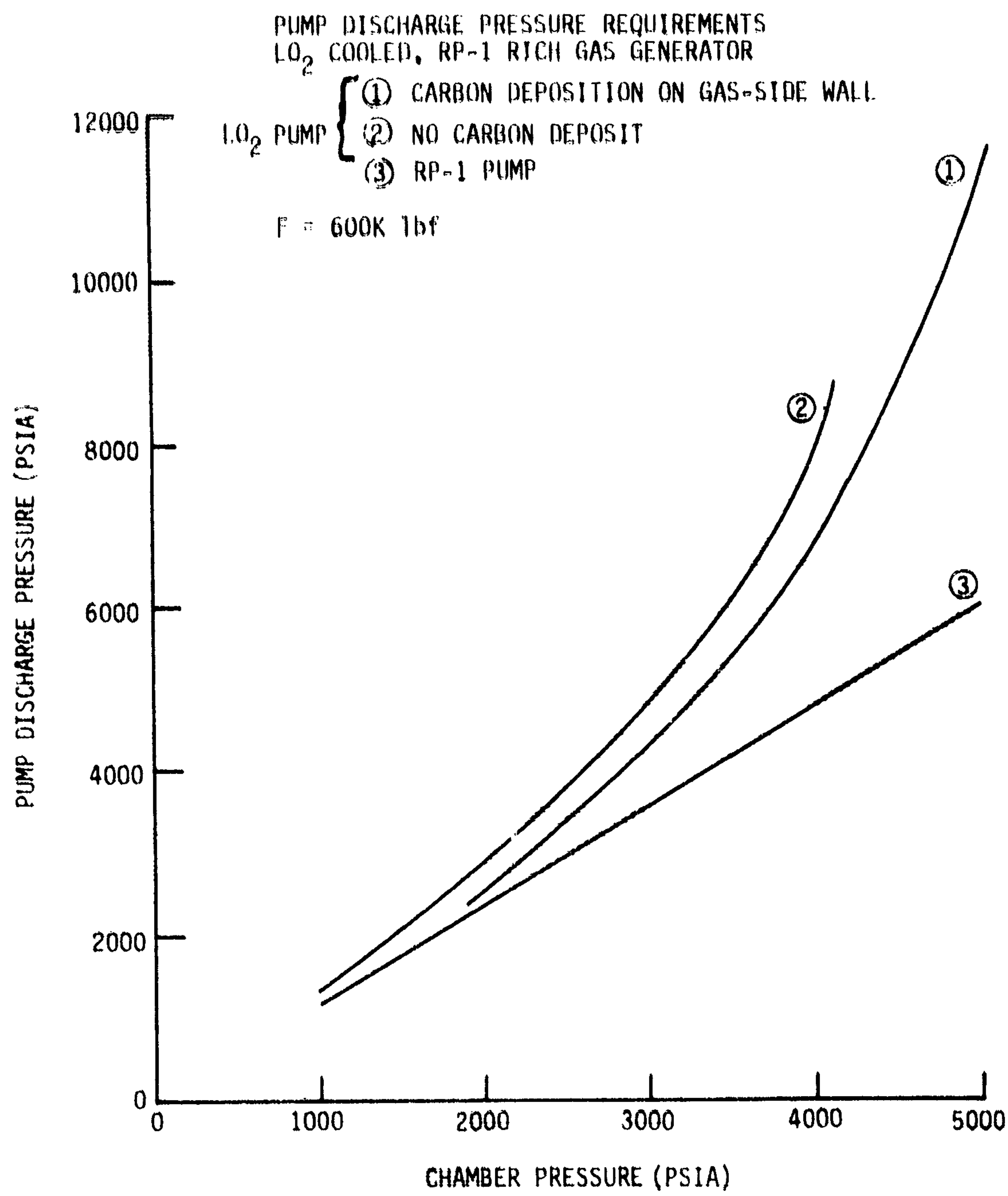



Figure 7. $LO_2$/RP-1 Engine Cycle B Pump Discharge Pressure Requirements

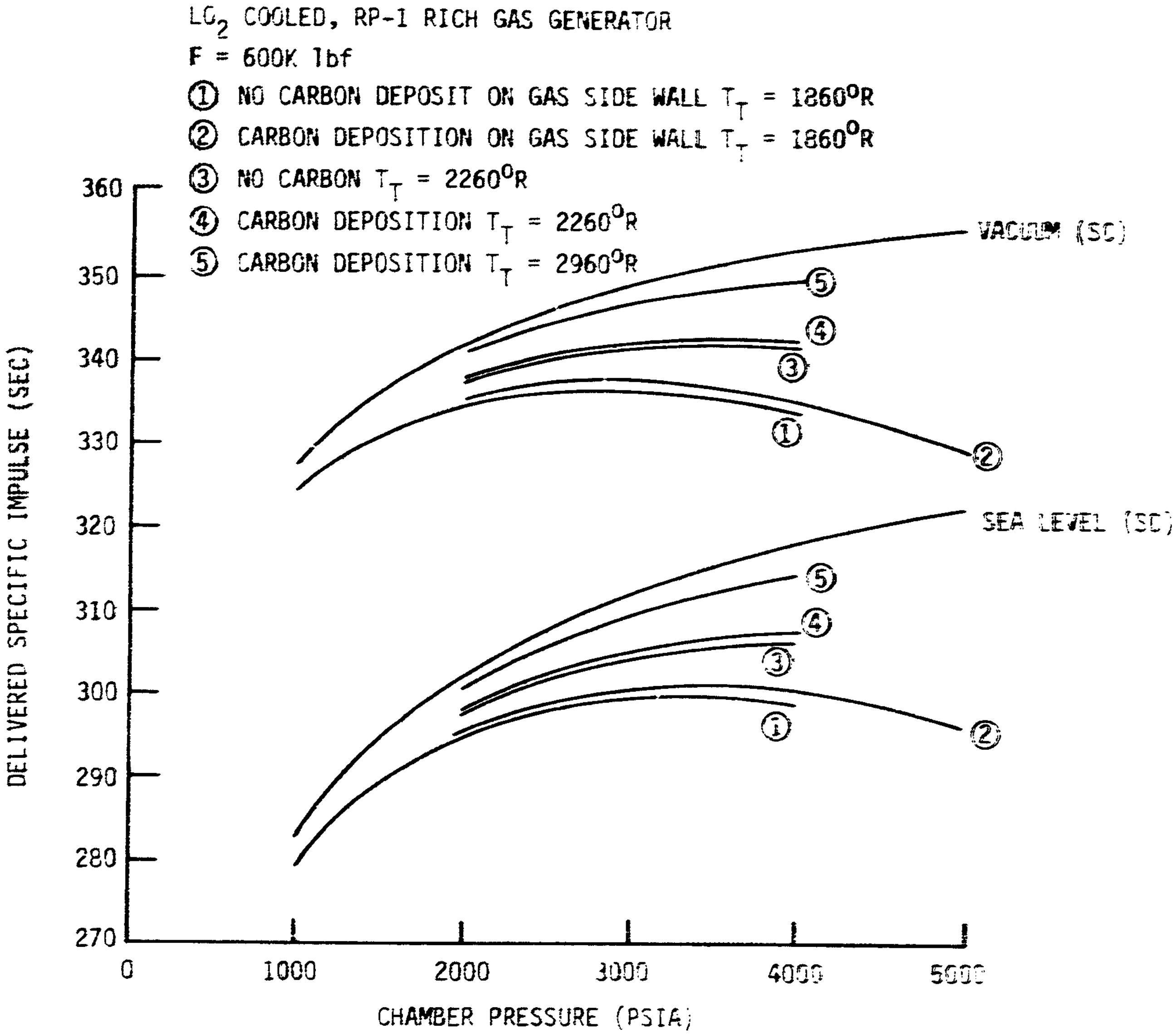



Figure 8. $LO_2$/RP-1 Engine Cycle B Performance

shows a large increase in performance (from 4 to 12 seconds). It should also be noticed that the uncoated chamber reaches its maximum performance at a chamber pressure of about 3000 psia. A carbon deposit and/or an increase in turbine inlet temperature are seen to shift the maximum performance to higher chamber pressures. The same trend is indicated in Figure 5 for the RP-1 cooled gas generator cycle engine.

c. Cycle C

Power balance data for the methane cooled gas generator cycle shown in Figure 9 are summarized in Figures 10 and 11. For an assumed pump discharge pressure limit (1980 state-of-the-art) of 8000 psia, a methane-cooled gas generator cycle engine is limited to a chamber pressure of 4300 psia. A carbon deposit on the chamber wall, as seen for cycles A and B, would allow an even higher chamber pressure.

Performance data for three turbine inlet temperatures are given in Figure 11. The initial increase in turbine inlet temperature from 1860 to 2260°R offers an increase in performance of about 4 seconds. Further increase in temperature to 2960°F is seen to give only about one second in additional engine performance.

d. Cycle D

The schematic for the $LO_2$/RP-1 staged combustion cycle D is given in Figure 12. The power balance data are summarized in Figure 13. Since the staged combustion cycle is a closed loop cycle, all its variations deliver the same performance. The performance data have previously been summarized (cf. Figures 5 and 8 and Table II).

The $LO_2$/RP-1, RP-1 cooled, staged combustion cycle is seen in Figure 13 to be limited to chamber pressures between 2500 and 3300 psia, if an upper limit of 8000 psia pump discharge pressure is assumed.

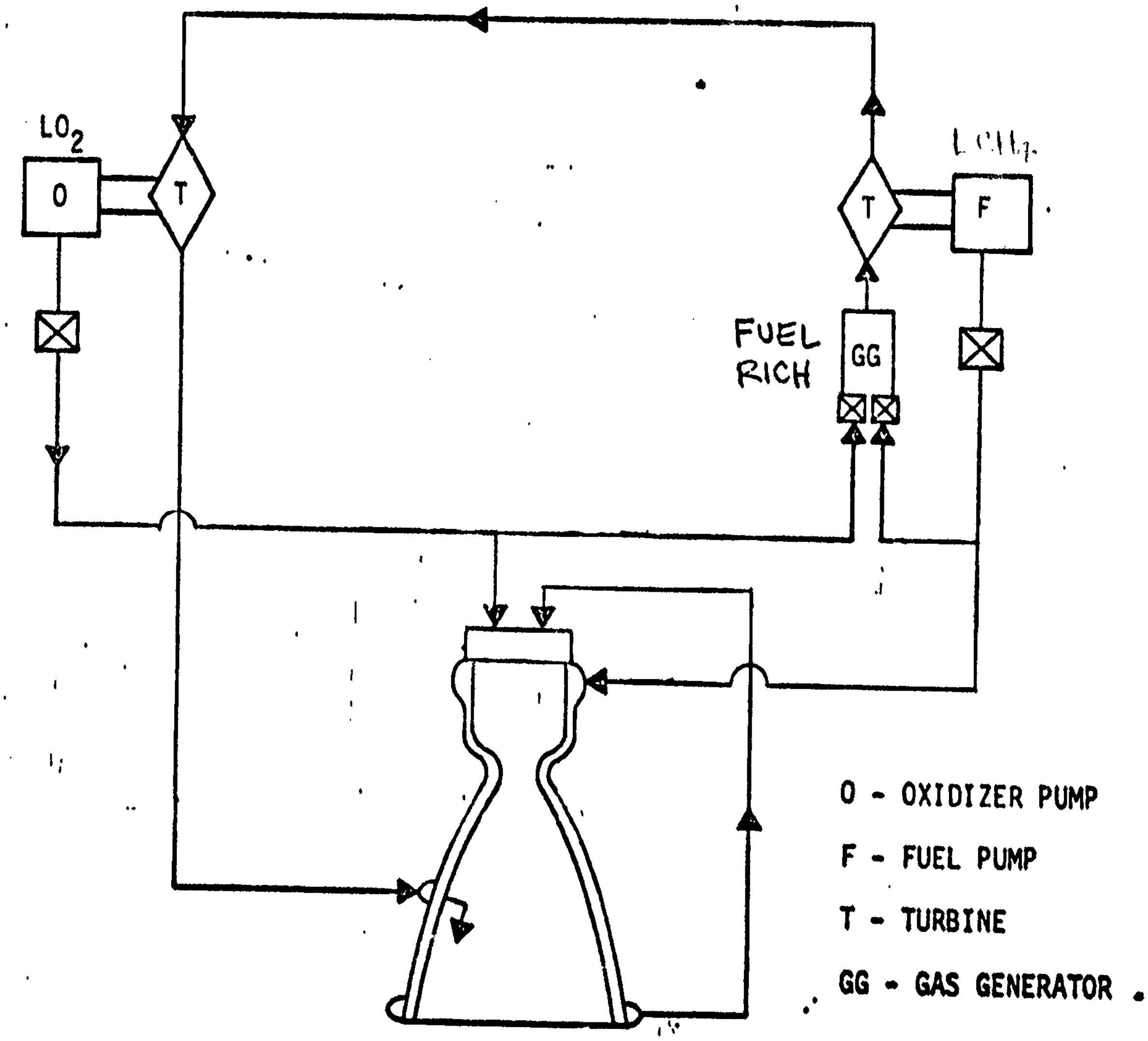



Figure 9. $LCH_4$ Fuel-Rich Gas Generator Cycle (C) $LCH_4$ Cooled

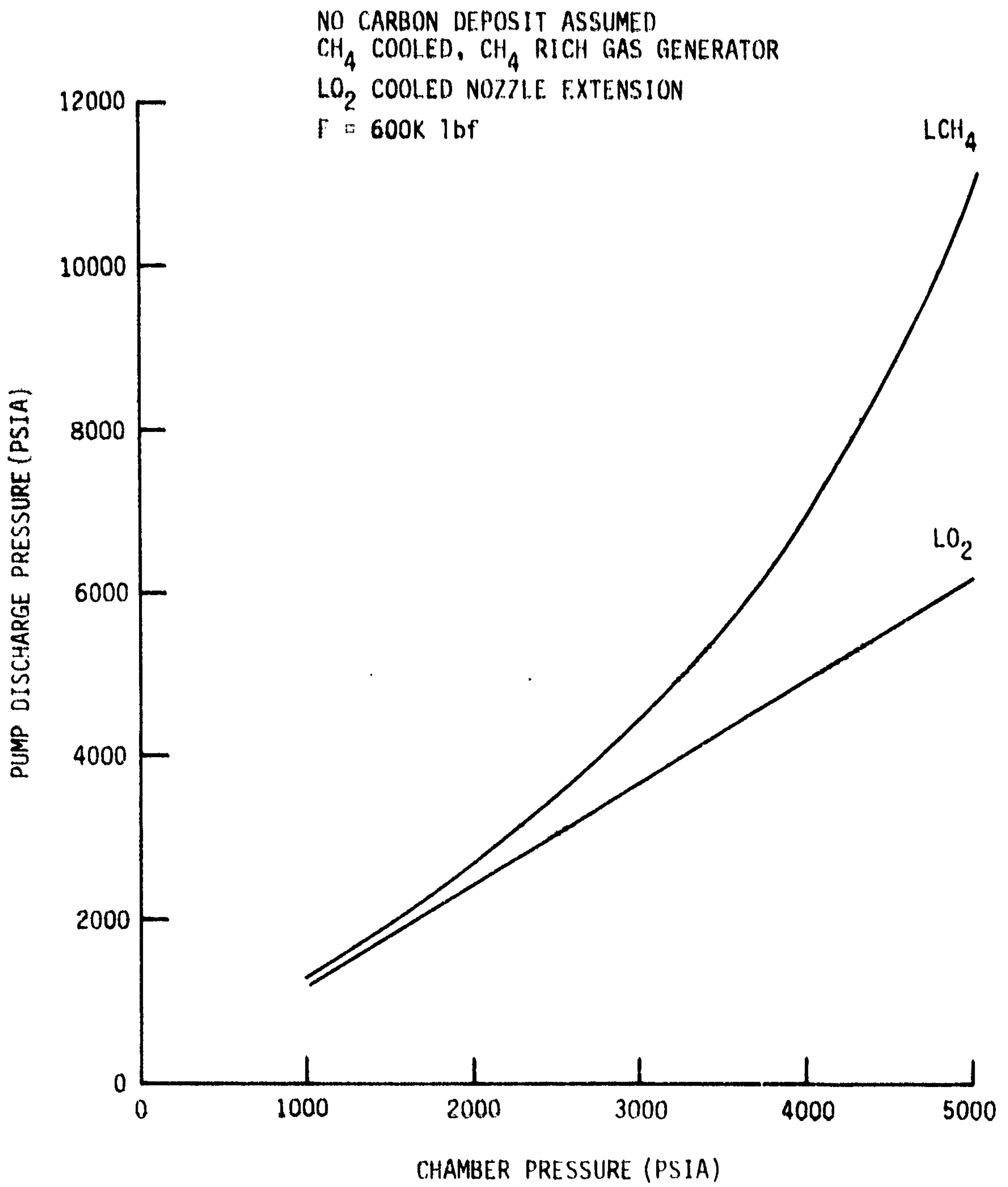



Figure 10. $LO_2/LCH_4$ Engine Cycle C Pump Discharge Pressure Requirements

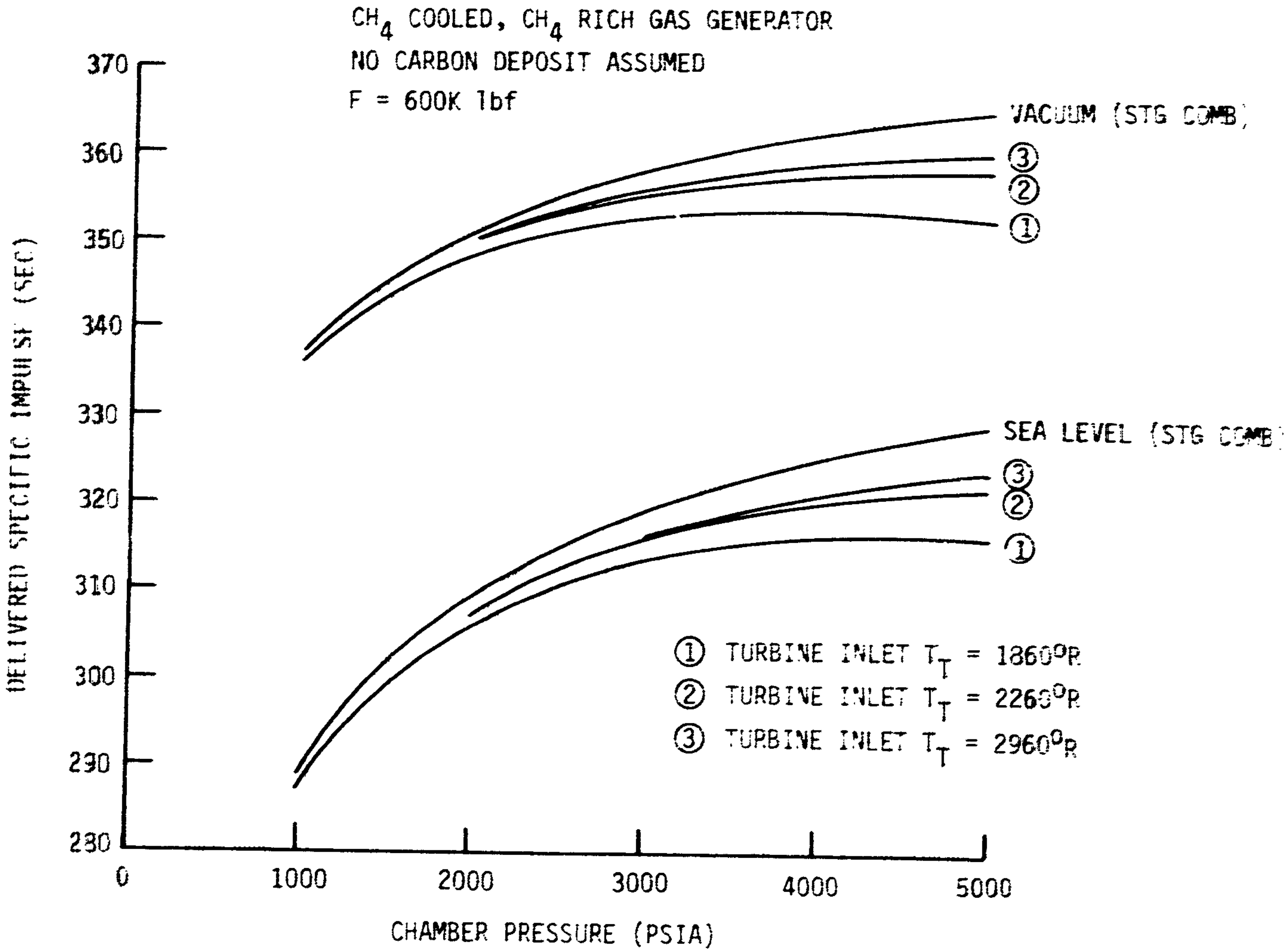



Figure 11. $LO_2/LCH_4$ Engine Cycle C Performance

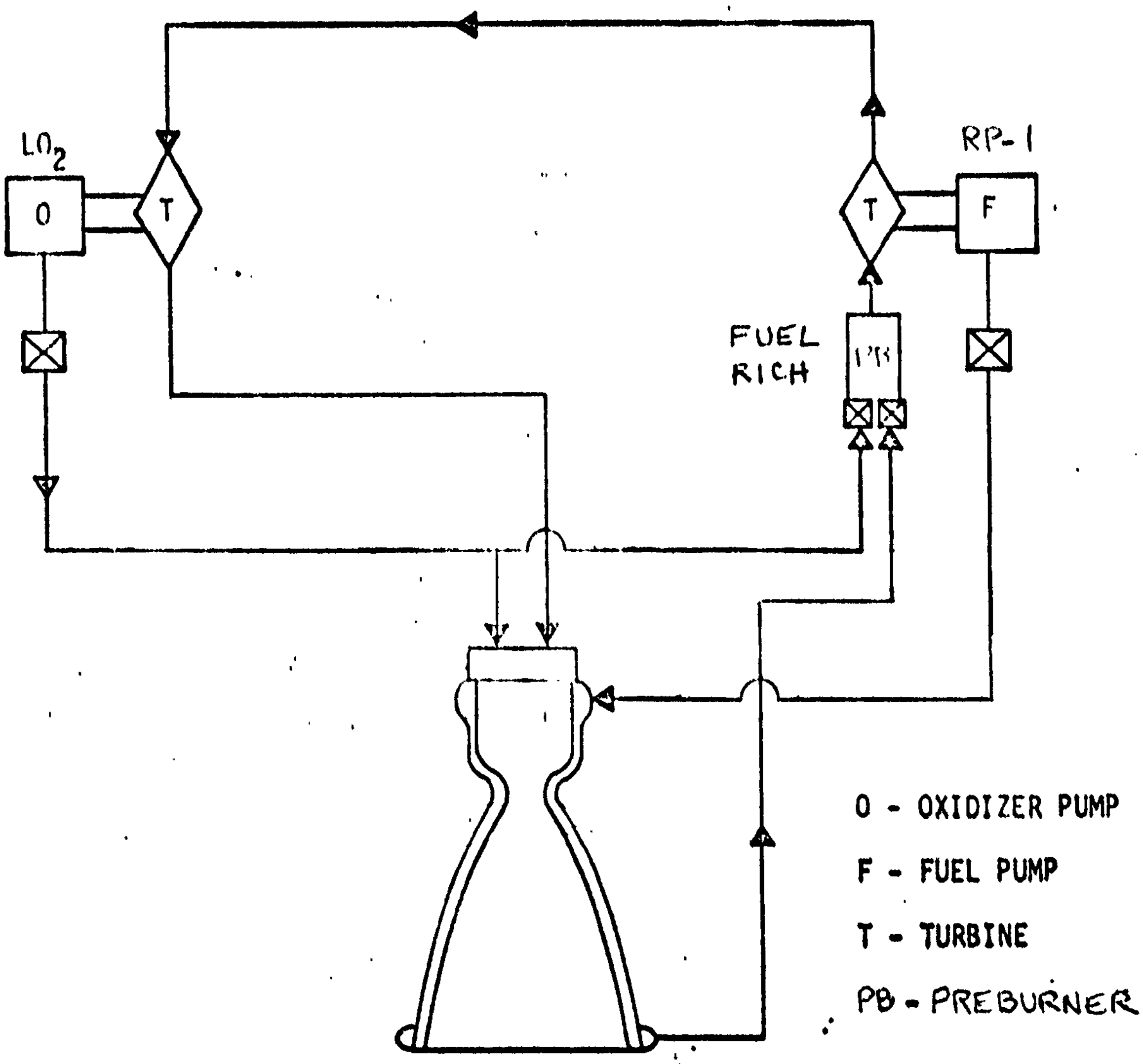



Figure 12. RP-1 Fuel-Rich Preburner Staged Combustion Cycle (D) RP-1 Cooled

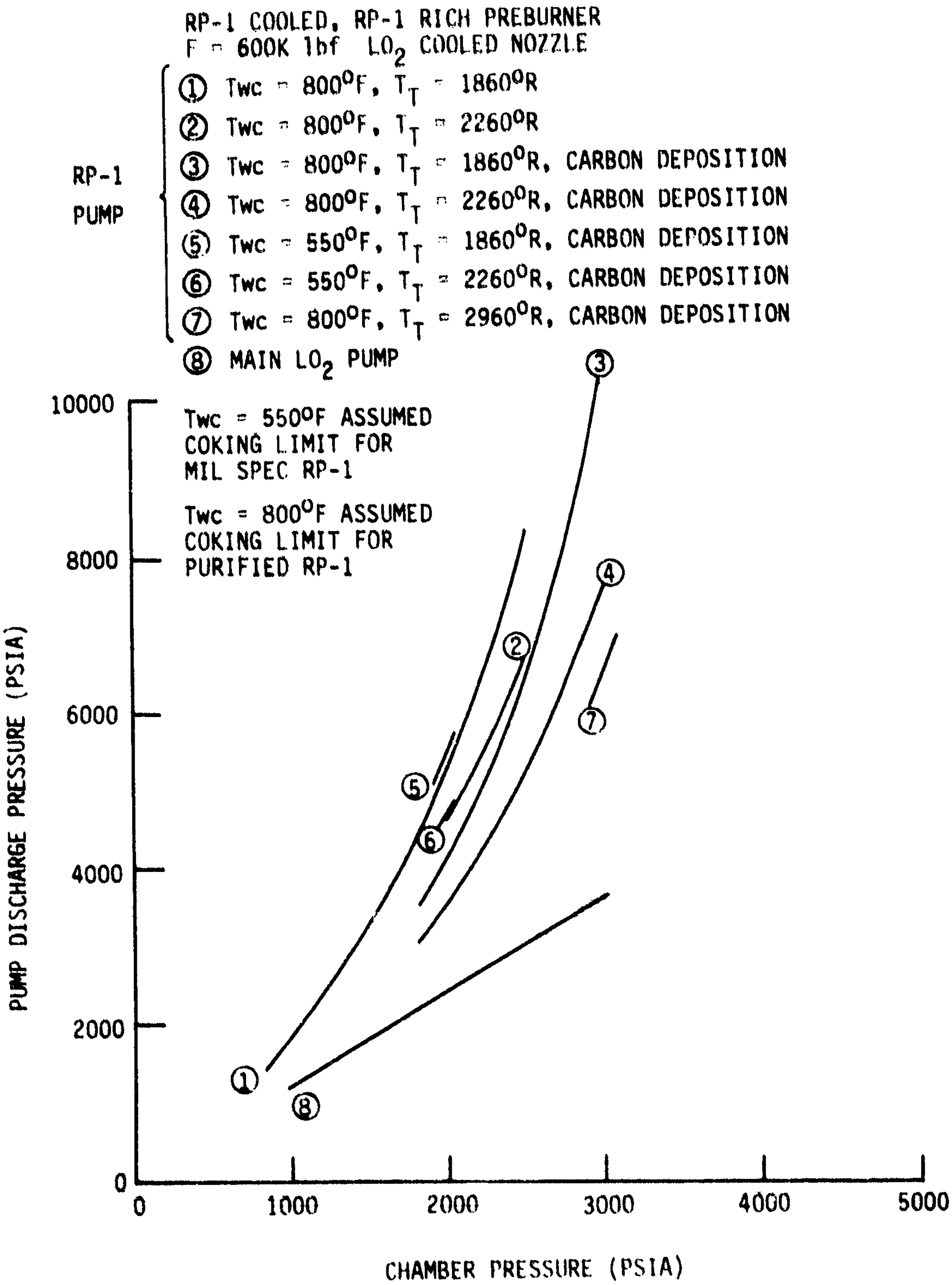



Figure 13. $LO_2$/RP-1 Engine Cycle D Pump Discharge Pressure Requirements

e. Cycle E

Cycle E (Figure 14) differs from cycle D by utilizing $LO_2$ as the coolant rather than RP-1 or purified RP-1. A modest increase in chamber pressure from 2500 (curve 1, Figure 13) to 2900 psia (curve 1, Figure 15), is achieved by changing coolants. The effects of carbon deposit and turbine inlet temperature are also indicated in Figure 15.

f. Cycle F

Cycle F differs from cycle D in the use of a $LO_2$ rich preburner in place of the RP-1 rich preburner. The cycle schematic is depicted in Figure 16 and the power balance summary is given in Figure 17. The required pump discharge pressures for the three staged combustion cycles (cycles D, E and F) at a chamber pressure of 2500 psia are:

| Cycle | $P_D$ | Coolant | Preburner |
|---|---|---|---|
| D | 8200 | RP-1 | RP-1 rich |
| E | 5800 | $LO_2$ | RP-1 rich |
| F | 5100 | RP-1 | $LO_2$ rich |

The $LO_2$ rich preburner because of its high mass flow provides more horsepower resulting in a lower pump discharge pressure requirement.

The influence of carbon deposit and turbine inlet temperature were not computed for this cycle. The effect of these variables should be similar to that previously shown.

g. Cycle G

The utilization of both a $LO_2$ rich preburner and $LO_2$ cooling is indicated in the schematic (Figure 18) for Cycle G, $LO_2$/RP-1 staged combustion cycle. Figure 19 presents the power balance results for this cycle. At a chamber pressure of 2500 psia, the pump discharge pressure requirement is 5200 psia (similar to that for cycle F). The maximum chamber pressure allowed by this cycle is 3150 psia if a pump discharge limit of 8000 psia (1980 state-of-the-art) is assumed.

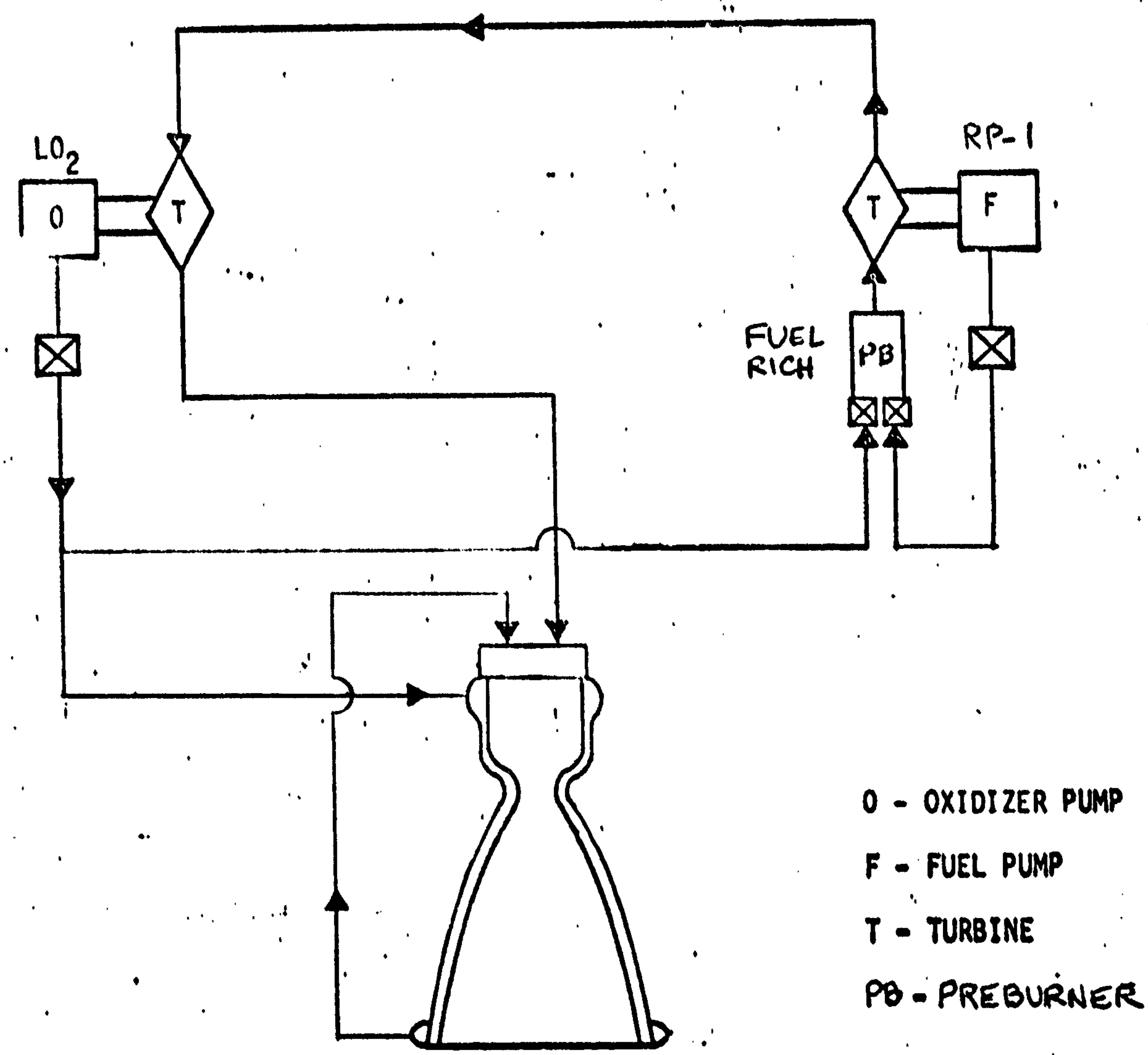



Figure 14. RP-1 Fuel-Rich Preburner Staged Combustion Cycle (E) $LO_2$ Cooled

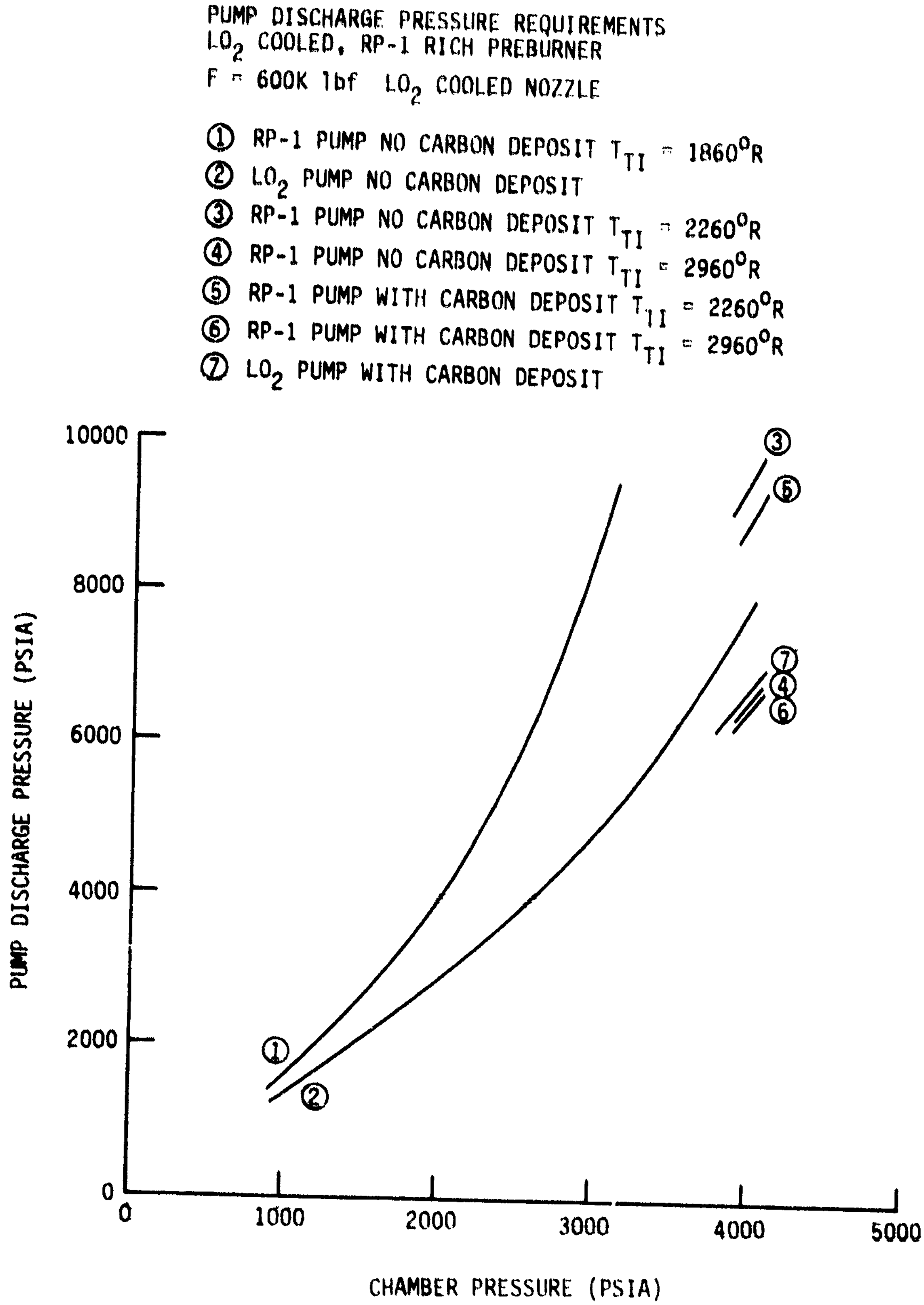



Figure 15. $LO_2$/RP-1 Engine Cycle E Pump Discharge Pressure Requirements

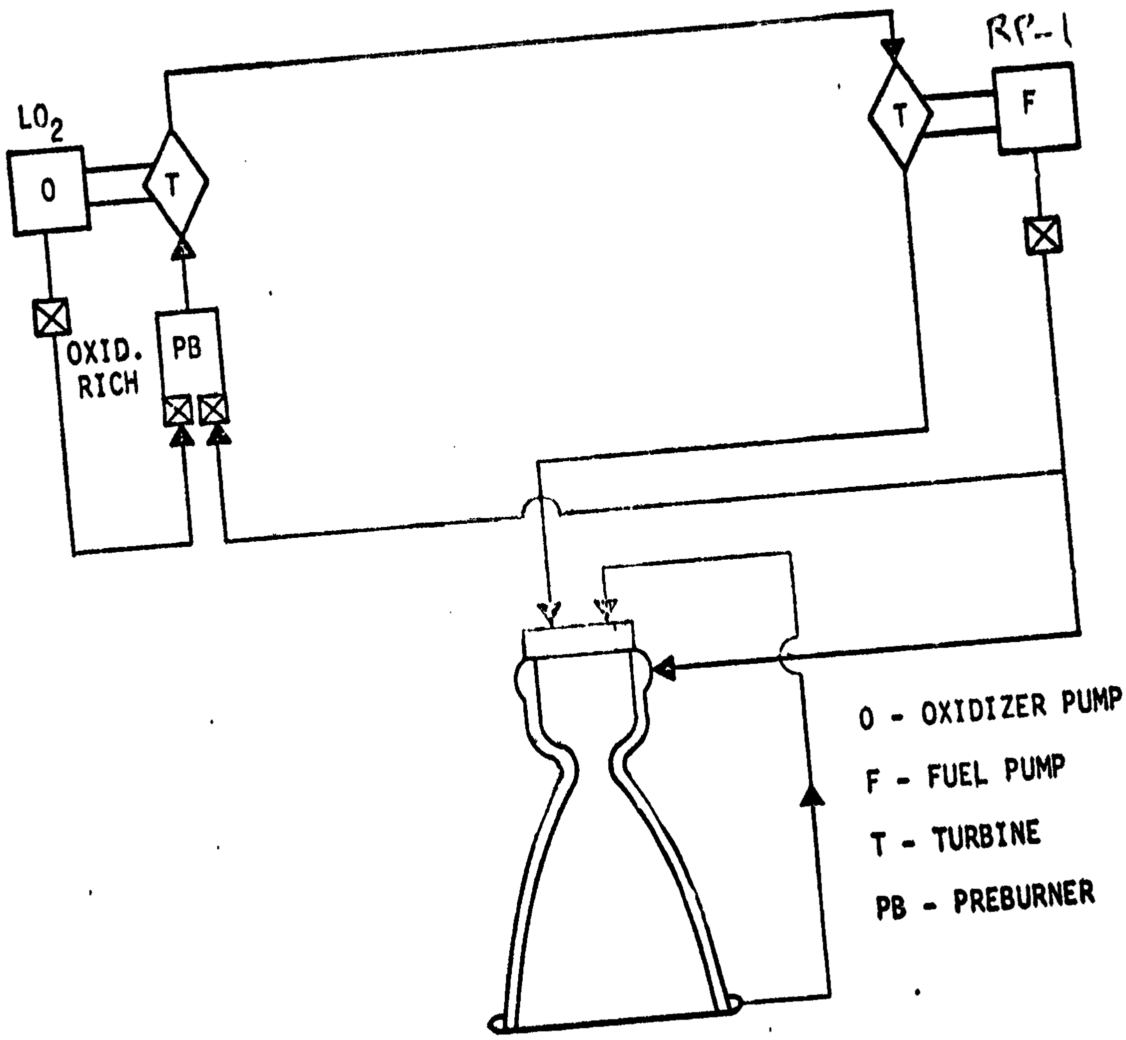



Figure 16. $LO_2$/RP-1 Oxidizer-Rich Preburner Staged Combustion Cycle (F) RP-1 Cooled

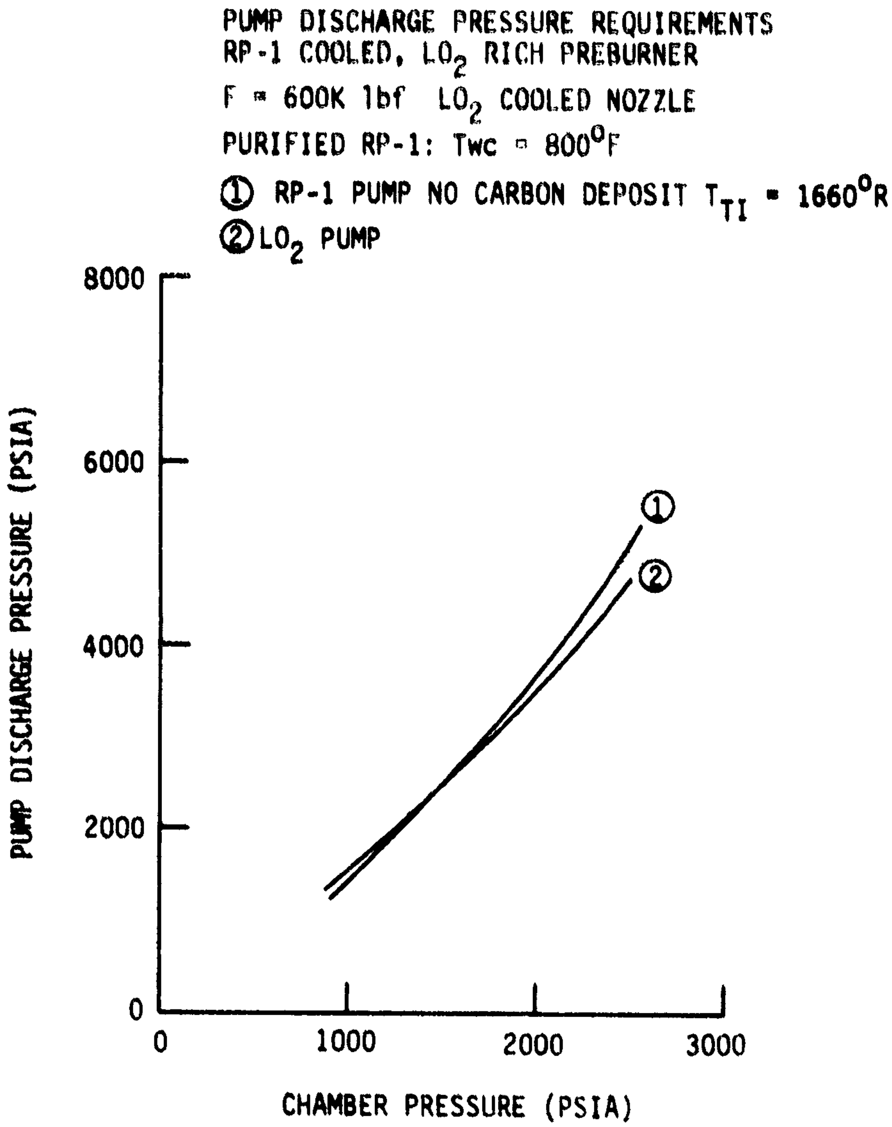



Figure 17. $LO_2$/RP-1 Engine Cycle F Pump Discharge Pressure Requirements

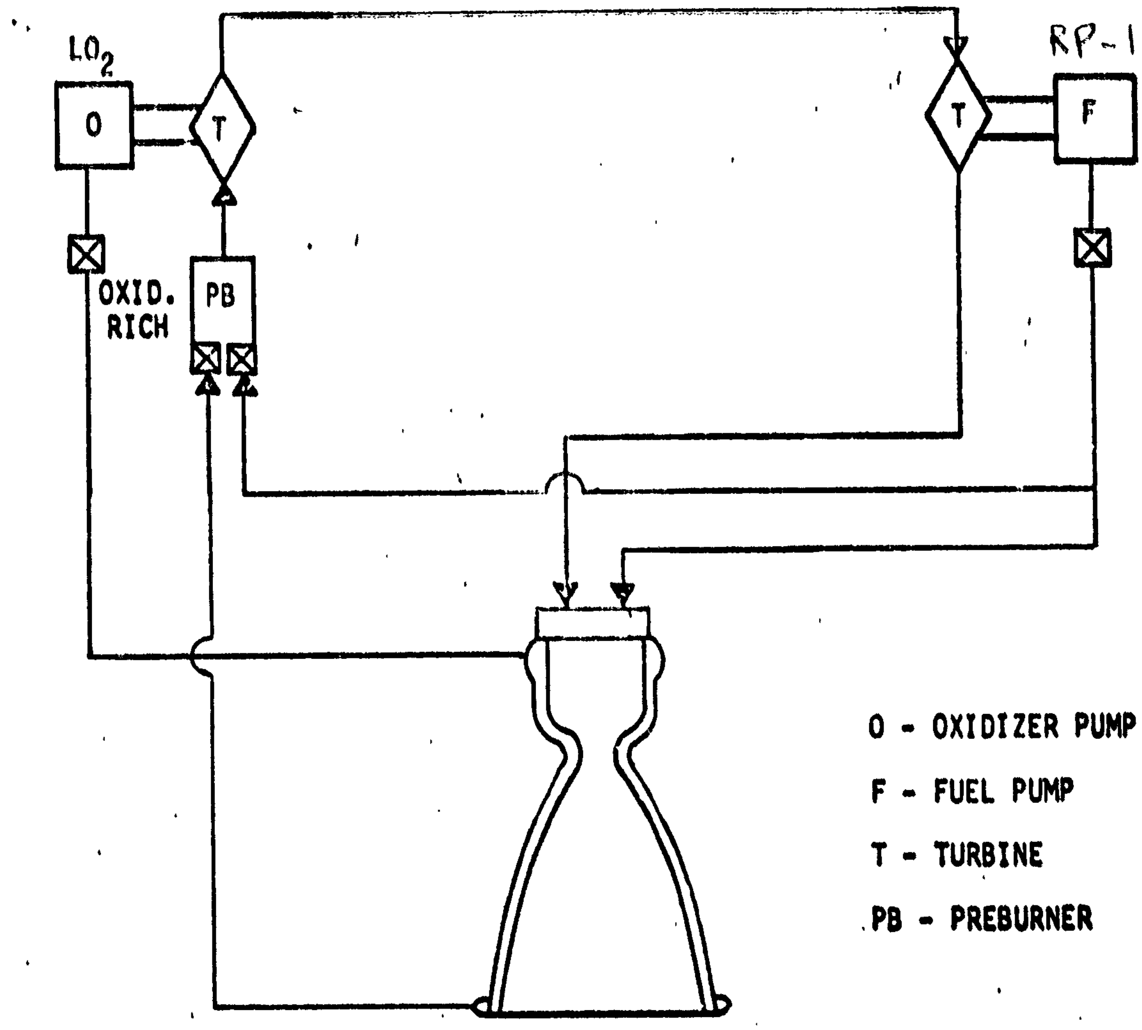



Figure 18. $LO_2$/RP-1 Oxidizer-Rich Preburner Staged Combustion Cycle (G) $LO_2$ Cooled

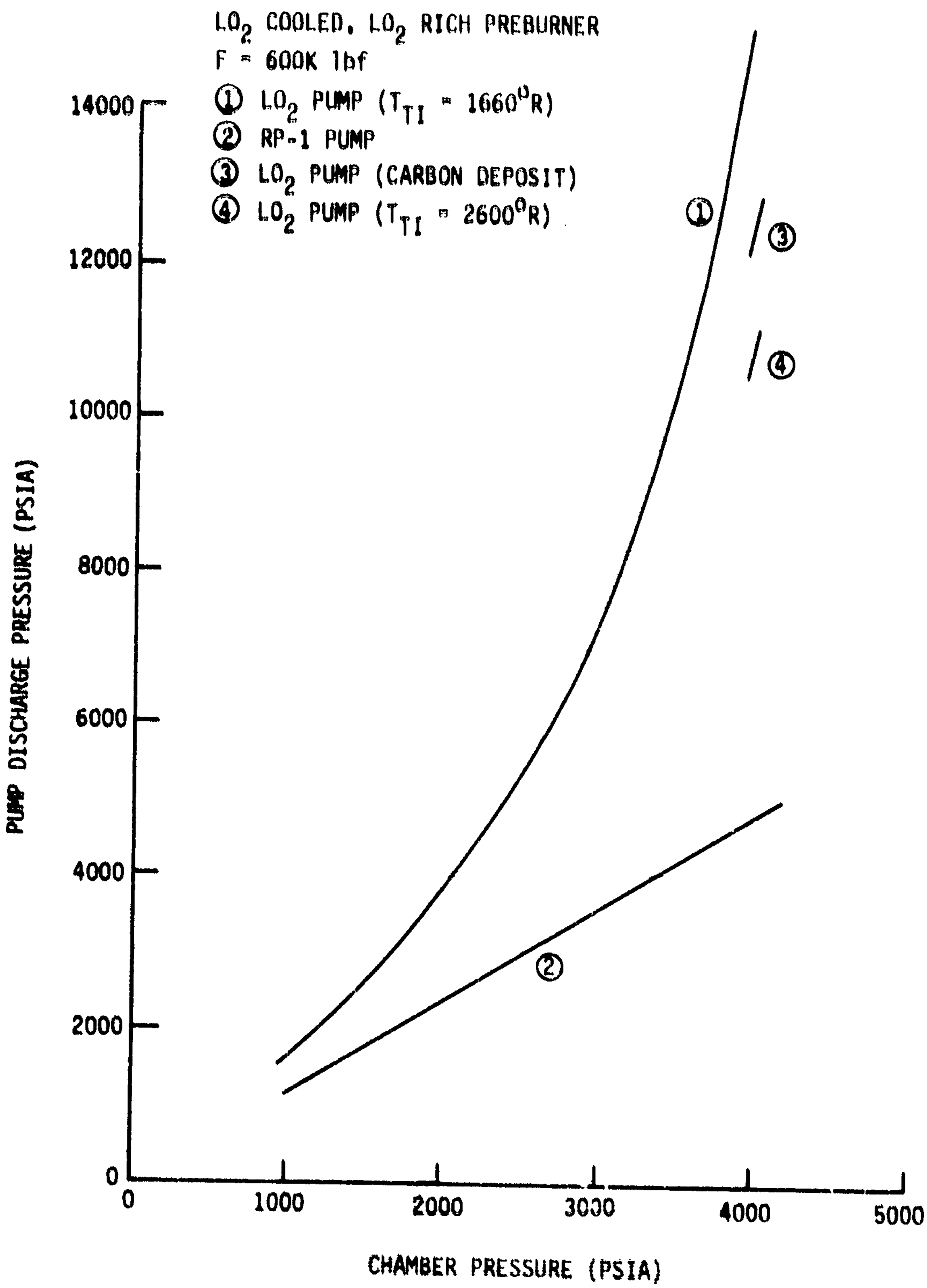



Figure 19. $LO_2$/RP-1 Engine Cycle G Pump Discharge Pressure Requirements

The effect of a chamber wall carbon deposit and of a higher turbine inlet temperature are also shown in Figure 19 at a chamber pressure of 4000 psia. The state-of-the-art turbine inlet temperature for rocket engine oxidizer-rich preburners is 1218°F (1678°R) at a pressure of 4556 psia based on the ARES program (Beichel, R., "Advanced Rocket Engine - Storable", Aerojet-General Corporation Report AFRPL-TR-67-75, Contract AF04(611)-10830, August 1967). However, an advanced ARES program (Kuntz, R.J., Sjogren, R.G., et al., "Advanced Propellant Staged-Combustion Feasibility Program", Aerojet-General Corporation Report AFRPL-TR-67-204, Contract AF04(611)-10785, September 1967) utilized an oxidizer-rich monopropellant (98% $H_2O_2$) preburner (no turbine) operating at 4500 psia and 1780°F (2240°R). Since the upper limit of feasible oxidizer-rich turbine-inlet temperatures has not been established, a temperature of 2600°R was selected for the one example shown in Figure 19.

The conclusions to be made concerning staged combustion cycles D through G are: (1) an oxidizer rich preburner offers a significant improvement (lower pump discharge pressures lead to longer life turbopumps); (2) $LO_2$ cooling significantly reduces the pump discharge pressure requirements of a fuel-rich preburner cycle; and (3) higher turbine inlet temperatures can lead to a lower pump discharge pressure (longer life) and/or to a higher chamber pressure.

h. Cycle H

The schematic for the $LO_2/LCH_4$ staged combustion cycle H is shown in Figure 20. The schematic is identical to that for cycle D, with methane replacing RP-1. The results of the power balance analysis are summarized in Figure 21.

If a pump discharge pressure limit of 8000 psia is assumed to be 1980 state-of-the-art, the chamber pressure limit for cycle H is 3000 psia (see Figure 21). This limit is increased to 3800 psia if the turbine inlet temperature can be increased to 2500°F (2960°R).

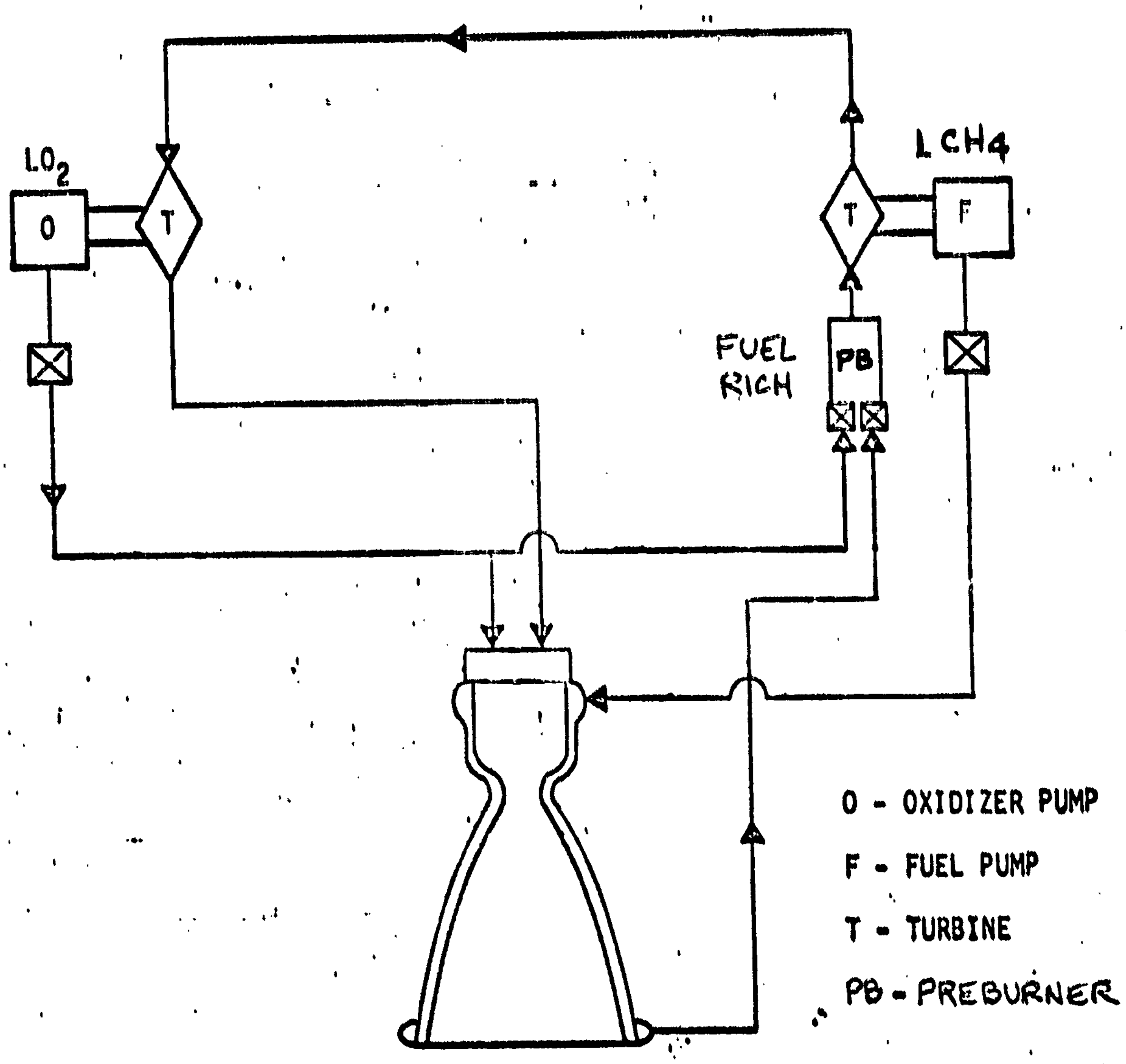



Figure 20. $LCH_4$ Fuel-Rich Preburner Staged Combustion Cycle (H) $LCH_4$ Cooled

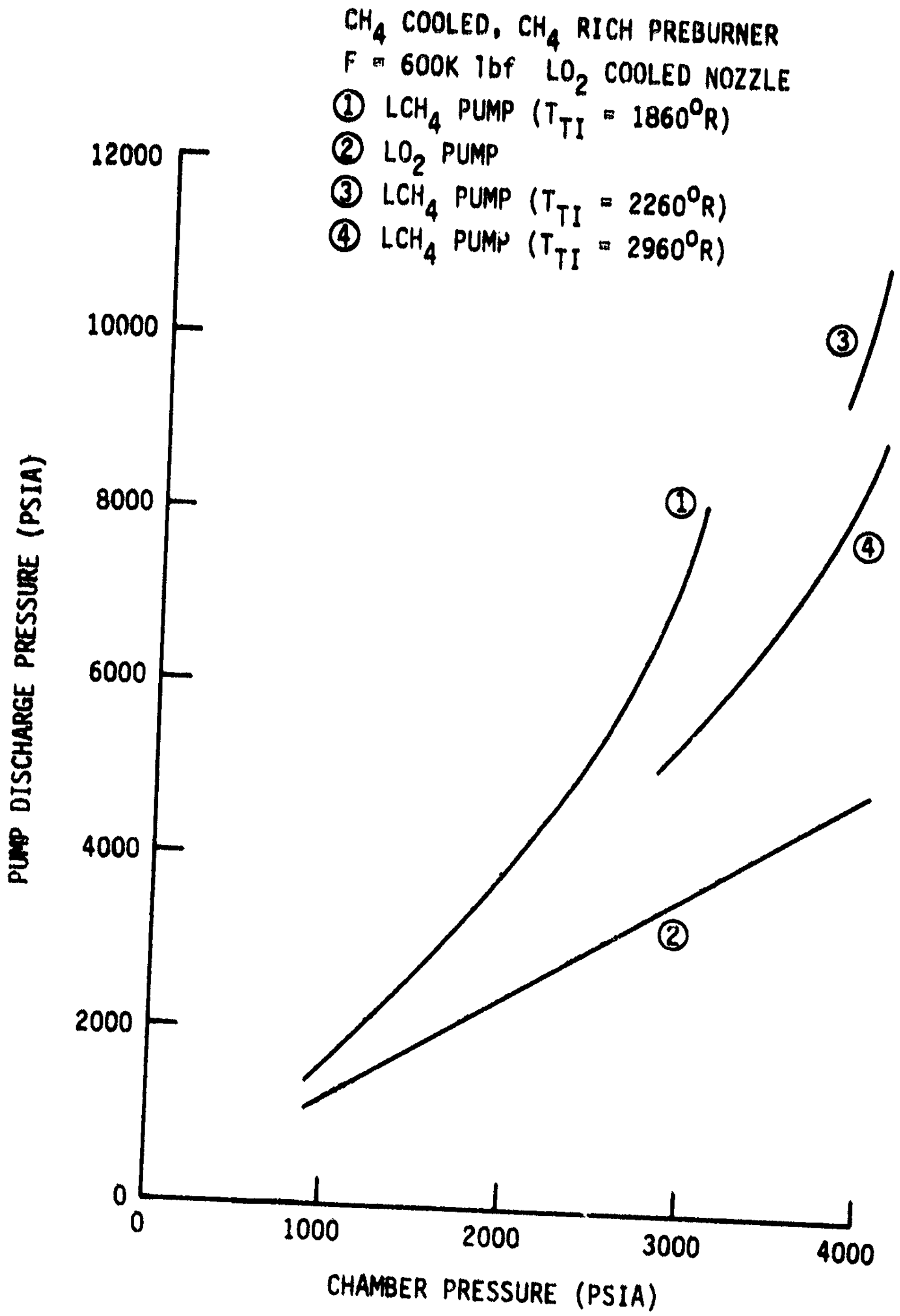



Figure 21. $LO_2/LCH_4$ Engine Cycle H Pump Discharge Pressure Requirements

No calculations were made for cycle H assuming a carbon deposit on the chamber wall. Although some deposit will probably exist, it is expected to be much lighter than that found from the combustion of $LO_2$ and RP-1.

i. Cycle I

Cycle I differs from cycle H in the addition of an oxidizer-rich preburner, as shown in Figure 22. The power balance data are summarized in Figure 23. The chamber pressure limit is seen to be 3500 psia (curve 1 in Figure 23) compared to 3000 psia (curve 1 in Figure 21). The benefit of the addition of an oxidizer-rich preburner to cycle H is directly translatable into a performance increase of 3.2 seconds (sea level) and 2.3 seconds (vacuum) because of the chamber pressure increase to 3500 psia.

The influence of turbine inlet temperature is indicated in the figure at a chamber pressure of 4000 psia. A large increase in the fuel-rich turbine inlet temperature (to 2960°R) significantly lowers the fuel pump discharge pressure. However, if the oxidizer-rich turbine inlet temperature is maintained constant at 1660°R (as shown in Figure 23), the reduction in flow through the oxidizer-rich turbine results in a higher $LO_2$ pump discharge pressure (curve 6) corresponding to the lower fuel discharge pressure (curve 5). A modest increase in fuel-rich turbine inlet temperature (curves 3 and 4) is, therefore, preferable in this case.

j. Cycle J

The schematic for a $LH_2$ cooled, $LH_2$ fuel-rich gas generator, $LO_2$/RP-1 engine cycle is depicted in Figure 24. The results from the power balance analysis for this cycle are summarized in Figures 25 and 26.

Cycle J is capable of generating a chamber pressure of 5000 psia at a pump discharge (6700 psia) well below the 8000 psia 1980 state-of-the-art value. The delivered performance for the engine slightly exceeds that for a staged combustion $LO_2$/RP-1 engine because of the addition of the $H_2$ fuel-rich turbine exhaust in the thrust chamber nozzle.

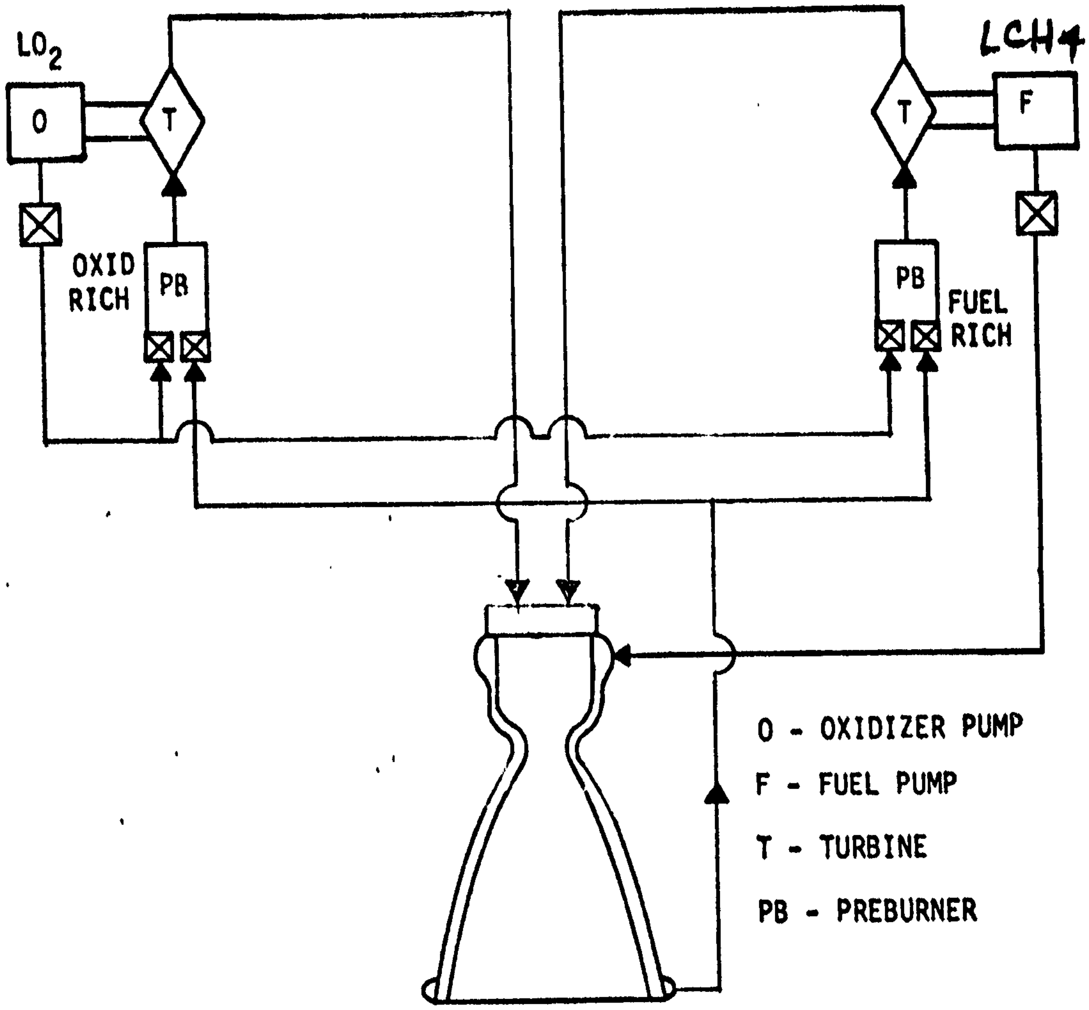



Figure 22. $LCH_4$ Mixed Preburner Staged Combustion Cycle (I) $LCH_4$ Cooled

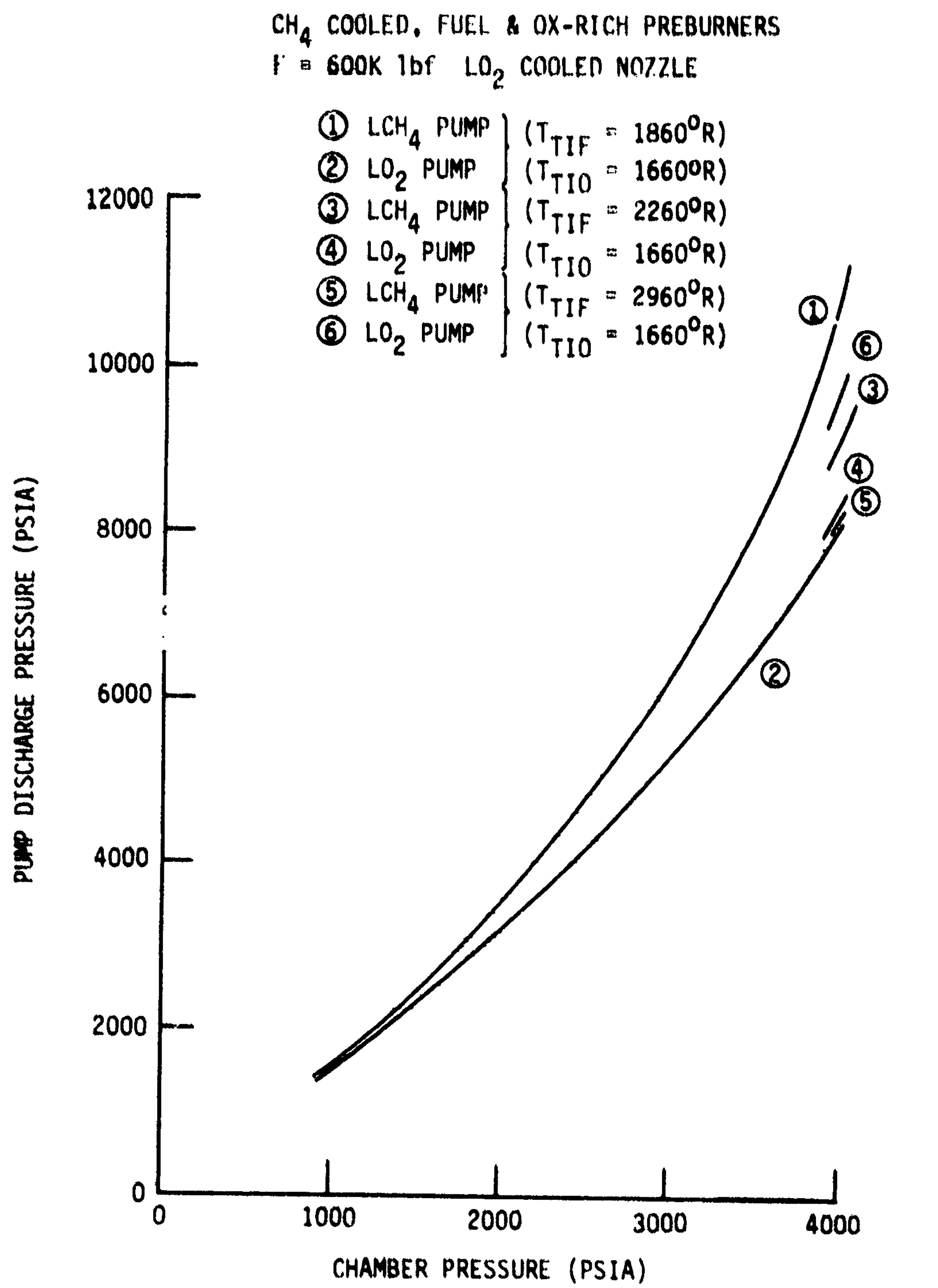



Figure 23. $LO_2/LCH_4$ Engine Cycle I Pump Discharge Pressure Requirements

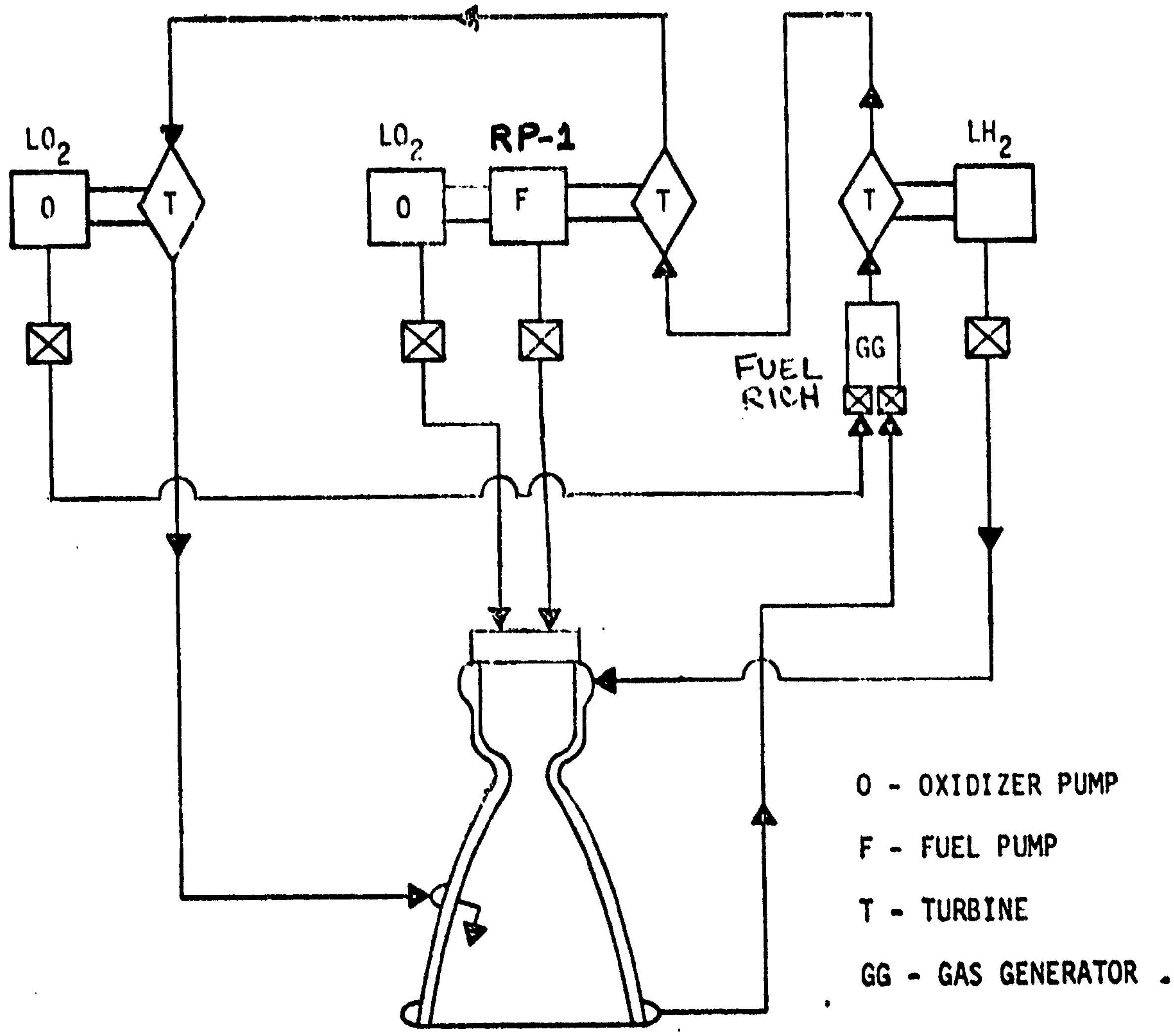



Figure 24. $LO_2$/RP-1 Engine Fuel-Rich $LH_2$ Gas Generator Cycle (J) $LH_2$ Cooled

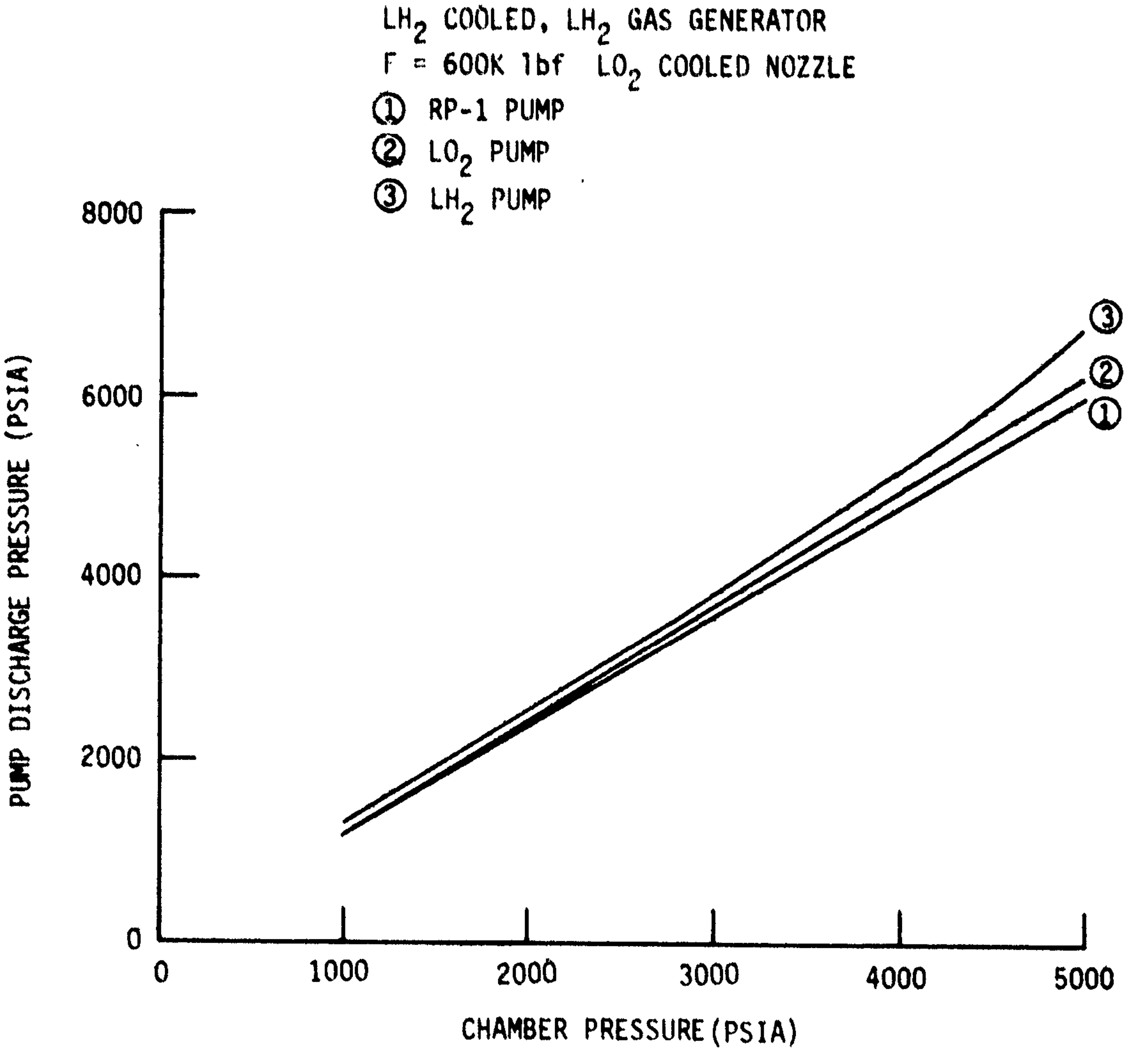



Figure 25. $LO_2$/RP-1 + $LH_2$ Engine Cycle J Pump Discharge Pressure Requirements

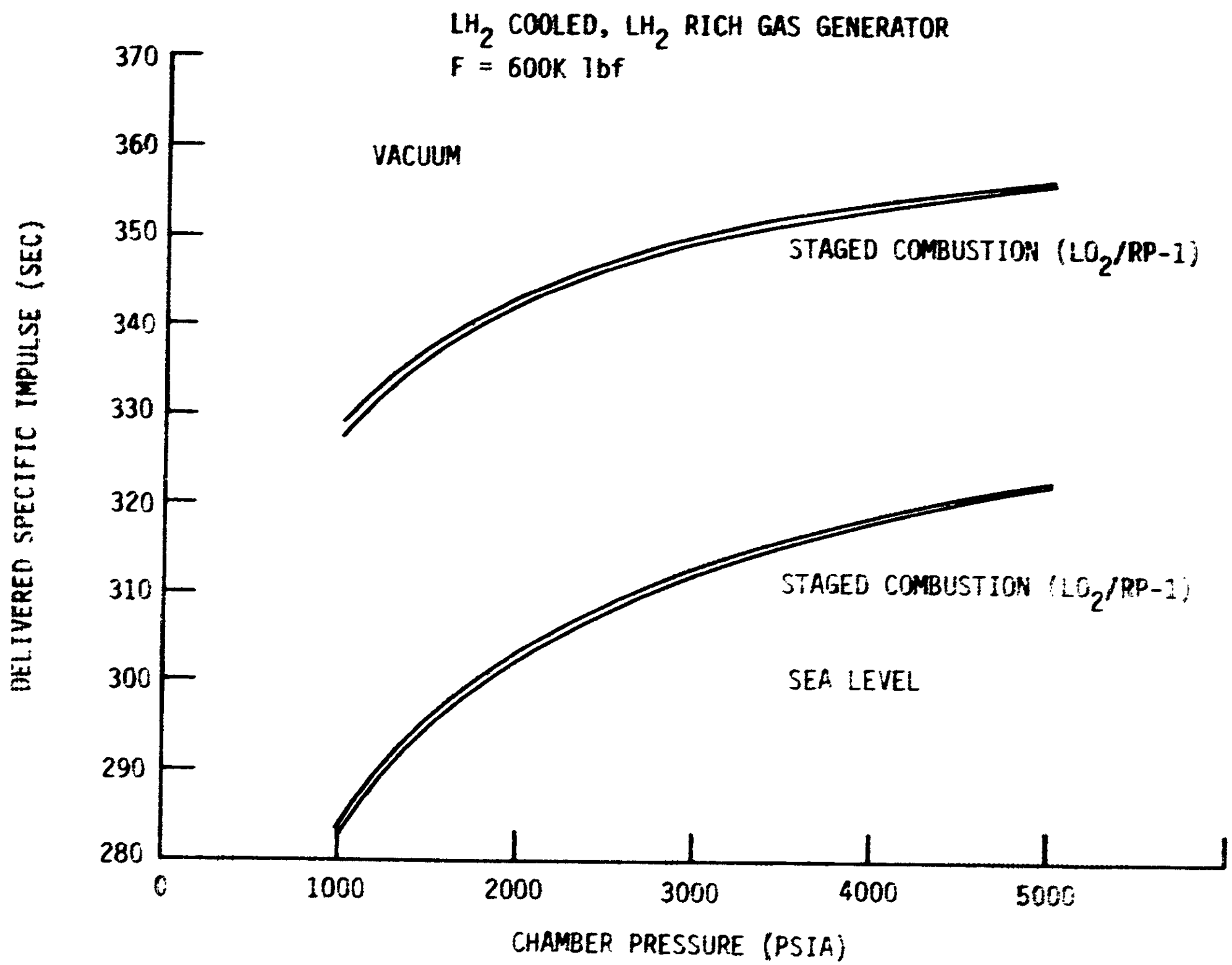



Figure 26. $LO_2$/RP-1 + $LH_2$ Engine Cycle J Performance

k. Cycle K

A $LO_2/LCH_4$ dual-throat engine schematic is shown in Figure 27. This engine utilizes both $LH_2$ and $LCH_4$ as coolants and both an oxidizer-rich preburner and a $H_2$ fuel-rich gas generator. The cycle shown in the schematic is representative of this class of engines, but a detailed heat transfer and thrust split analysis is required to fully optimize this type of engine for a two-stage mission. Sufficient data exist for similar tripropellant engines (O'Brien, C.J., "Dual-Fuel, Dual-Throat Engine Preliminary Analysis," Aerojet Liquid Rocket Company Report 32967F, Contract NAS 8-32967, August 1979) to allow power balance and performance analysis of this bipropellant engine with a hydrogen-rich gas generator drive. The specification for cycle K is given in Table V.

1. Thrust Level Variation

The parametric heat transfer data generated in Task I and the parametric performance data generated in Task II show some variation with thrust level from 200,000 to 1,500,000 lbf. Some of this variation is real and some is the result of approximations used in the parametric scaling relationships required to facilitate the generation of a wide variety of design data.

Past experience has shown that engine cycles can be rated at a given thrust level (e.g., 600,000 lbf) and that the rating will be valid for other thrust levels (i.e., 200,000 to 1,500,000 lbf). To validate this premise, power balance calculations were made for cycle C at thrust levels of 200,000, 600,000 and 1,500,000 lbf. The results are given in Table IV. The following table summarizes the pertinent data.

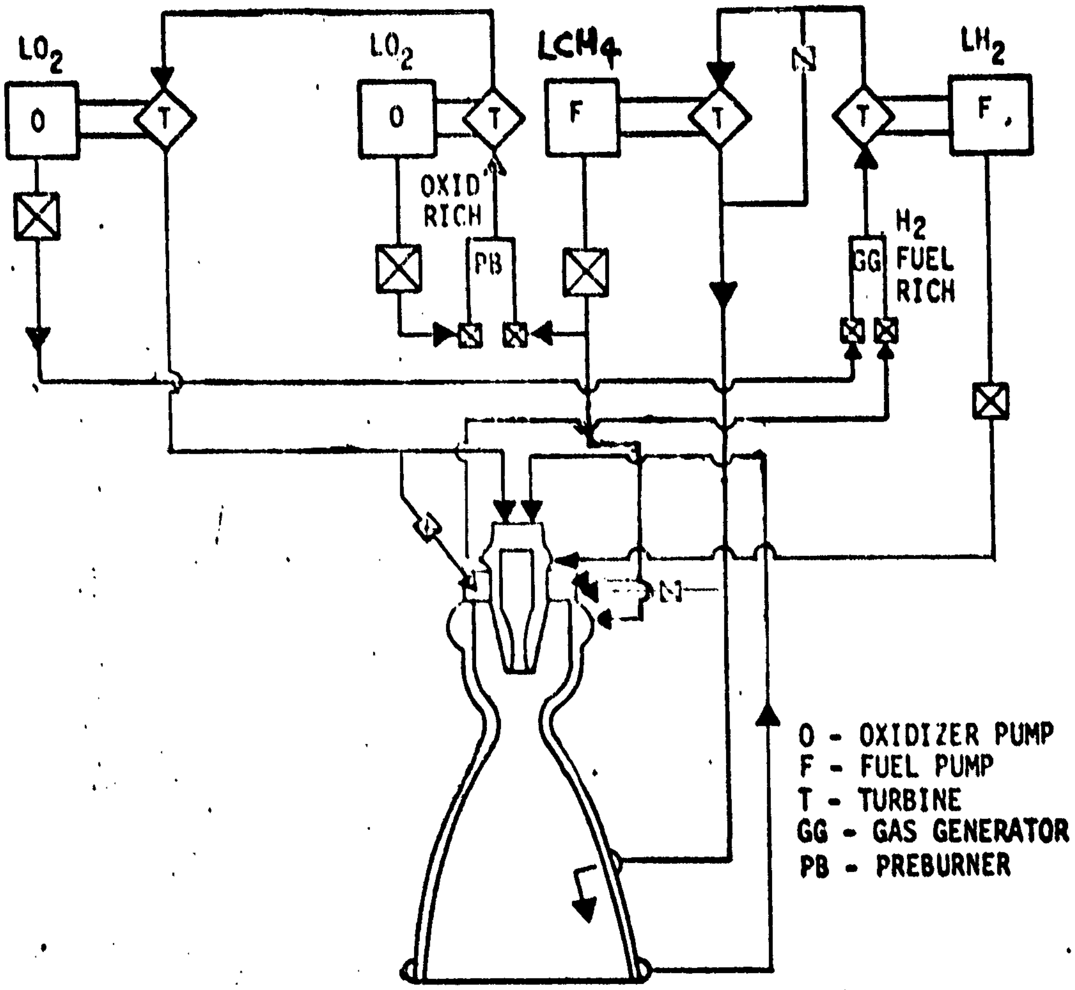



Figure 27. $LO_2/LCH_4$ Dual-Throat Engine Mixed Gas Generator/Staged Combustion Cycle (X) $LCH_4$ and $LH_2$ Cooled

| Thrust (lbf) | Chamber Pressure (psia) | Pump Dischg. Pressure (psia) | Sea Level Is (sec.) | Vacuum Is (sec.) |
|---|---|---|---|---|
| 200,000 | 4000 | 7030 | 315.8 | 353.5 |
| 600,000 | 4000 | 6921 | 316.5 | 354.1 |
| 1,500,000 | 4000 | 7662 | 316.5 | 354.1 |

It is seen that there is little variation in the performance of the engines over this wide range of thrust level. There is a variation in pump discharge pressure of the order of 10 percent, but some of this variation can be reduced through thrust chamber design changes to reduce the coolant pressure drop.

TABLE V

$LO_2/LCH_4$ ENGINE CYCLE K SPECIFICATION

| PARAMETER | MODE I | MODE II |
|---|---|---|
| Sea Level Thrust, LBF | 610,745 | -- |
| Vacuum Thrust, LBF | 685,270 | 225,960 |
| Sea Level IS, sec | 319.0 | -- |
| Vacuum Is, sec | 357.9 | 380.2 |
| Mixture Ratio ($LO_2/LCH_4$) | 3.5 | 3.5 |
| Mixture Ratio ($LO_2/LH_2$) | 0.8 | 0.8 |
| Chamber Pressure, psia | 2800/4000 | 4000 |
| Area Ratio | 42 | 187 |
| TCA Sea Level Is, sec | 320.9 | -- |
| TCA Vacuum Is, sec | 359.3 | 383.6 |
| GG Sea Level Is, sec | 238.8 | -- |
| GG Vacuum Is, sec | 300.7 | 338.1 |
| Flowrate, lb/sec | 1914.61 | 594.36 |
| TCA $LO_2$ Flowrate, lb/sec | 1454.14 | 427.28 |
| TCA $LCH_4$ Flowrate, lb/sec | 415.47 | 122.08 |
| GG $LO_2$ Flowrate, lb/sec | 20.00 | 20.00 |
| GG $LH_2$ Flowrate, lb/sec | 25.00 | 25.00 |
| Throat Area, $in^2$ | 115.22 | 26.06 |
| Exit Area, $in^2$ | 4869.72 | 4869.73 |
| Exit Pressure, psia | 7.3 | 1.5 |
| $LCH_4$ Pump Dischg. Pressure, psia | 7429 | 7429 |
| $LO_2$ Pump Dischg. Pressure, psia | 7429 | 7429 |
| $LH_2$ Pump Dischg. Pressure, psia | 1655 | 1655 |

2. Engine Cycle Rating System

Engine cycle rating parameters were established as shown in Table VI. The desired condition and the effect of the parameter on the engine and/or vehicle are listed in the table. Both low and high temperature turbines are listed as desirable, in view of the fact that this study indicates a significant benefit of a high turbine temperature on some cycles.

The chamber pressure ranking of the cycles is given in Figure 28. The ranking is based on an upper limit of pump discharge pressure of 8000 psia, the assumed 1980 state-of-the-art. The ranking is also based on 1980 state-of-the-art fuel-rich and oxidizer-rich turbine temperatures of 1860 and 1600 °R, respectively. The $LO_2$-, $LCH_4$-, and $LH_2$-cooled cycles (B,C,G,I,J and K) are seen to have the highest chamber pressure potential.

The chamber pressure ranking from Figure 28 can be converted into a performance ranking for the cycles, as shown in Figure 29. The performance values show less variation due to the high delivered performance for the staged combustion cycles, even at lower chamber pressures. The cycle with the highest performance potential is seen to be the dual throat cycle. The variable geometry (without moving parts) allows the achievement of a high performance at altitude.

Although preliminary designs are not available for each of the various cycles, an estimate of the component weight differences was made. These differences are reflected in the engine weight ranking depicted in Figure 30. Based on simplified trajectory analyses to be discussed in Section II.C these weight differences do not significantly influence the two-stage vehicle payload.

## TABLE VI

### ENGINE CYCLE RATING PARAMETERS

| PARAMETER | ENGINE CYCLE | | EFFECT |
|---|---|---|---|
| | SC | GG | |
| ENGINE PERFORMANCE | HIGH | LOWER | PAYLOAD CAPABILITY |
| ENGINE WEIGHT | HIGH | LOWER | PAYLOAD CAPABILITY |
| PUMP DISCHARGE PRESSURE | HIGH | LOWER | TPA CYCLE LIFE |
| CHAMBER PRESSURE | HIGH | LOWER | PAYLOAD CAPABILITY<br>ENGINE ENVELOPE |
| TANK MIXTURE RATIO | OPTIMUM | NOT OPTIMUM | PAYLOAD CAPABILITY |
| TURBINE TEMPERATURE | HIGH | LOWER | TPA CYCLE LIFE<br>COOLING SYSTEM COMPLEXITY |
| | HIGH | HIGH | POWER REQUIREMENT<br>PAYLOAD CAPABILITY |
| $LO_2$/HC FUEL-RICH TURBINE | COKING | COKING | TPA/INJECTOR CYCLE LIFE<br>PURGE SYSTEM REQUIREMENT<br>LOW PERFORMANCE |
| $LO_2/H_2$ FUEL-RICH TURBINE | LOW $P_2/P_1$ | HIGH $P_2/P_1$ | STAGED TURBINE<br>HIGH PERFORMANCE<br>LOW PUMP DISCHARGE PRESSURE |
| INTER-PROPELLANT SEAL REQUIREMENT FOR TPA | NO/YES | YES | TPA CYCLE LIFE & PERFORMANCE |
| COOLANT REQUIREMENT | HIGHER | LOWER | HOT GAS MANIFOLD COOLING IF TURBINE TEMP. >1200°F |

INDICATES DESIRABLE CONDITION

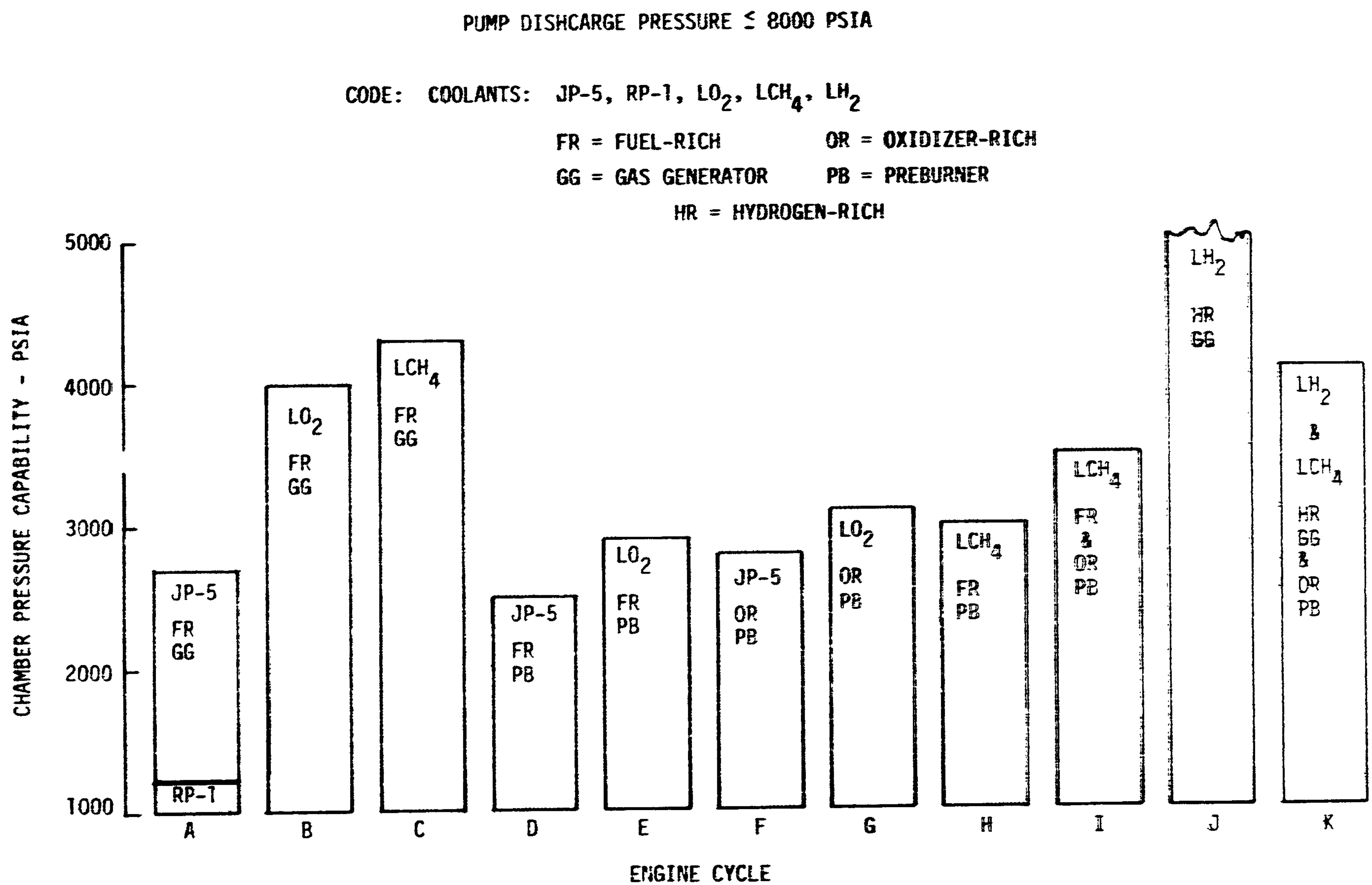



Figure 28. Chamber Pressure Ranking of $LO_2$/HC Engine Cycles

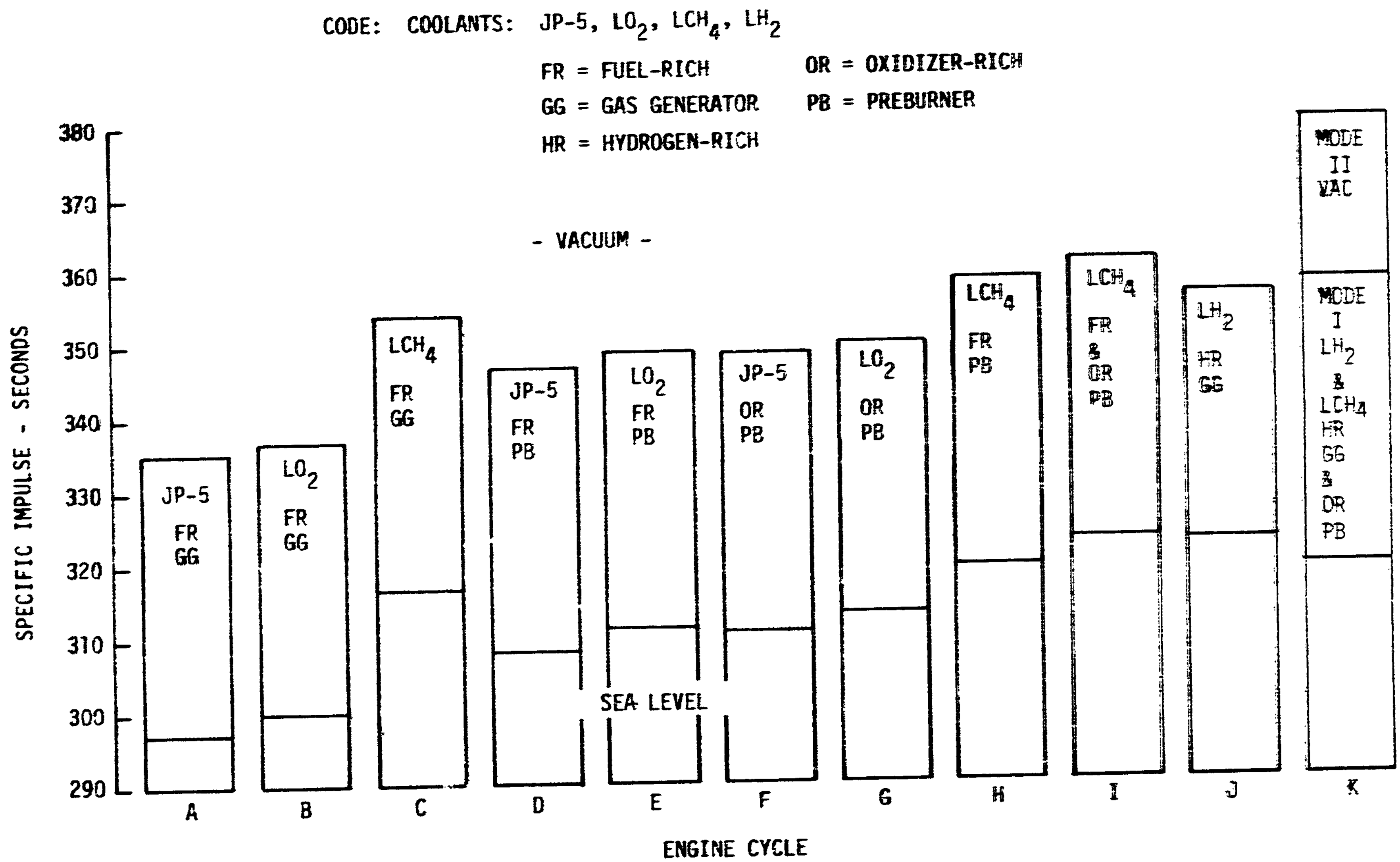



Figure 29. Performance Ranking of $LO_2$/HC Engine Cycles

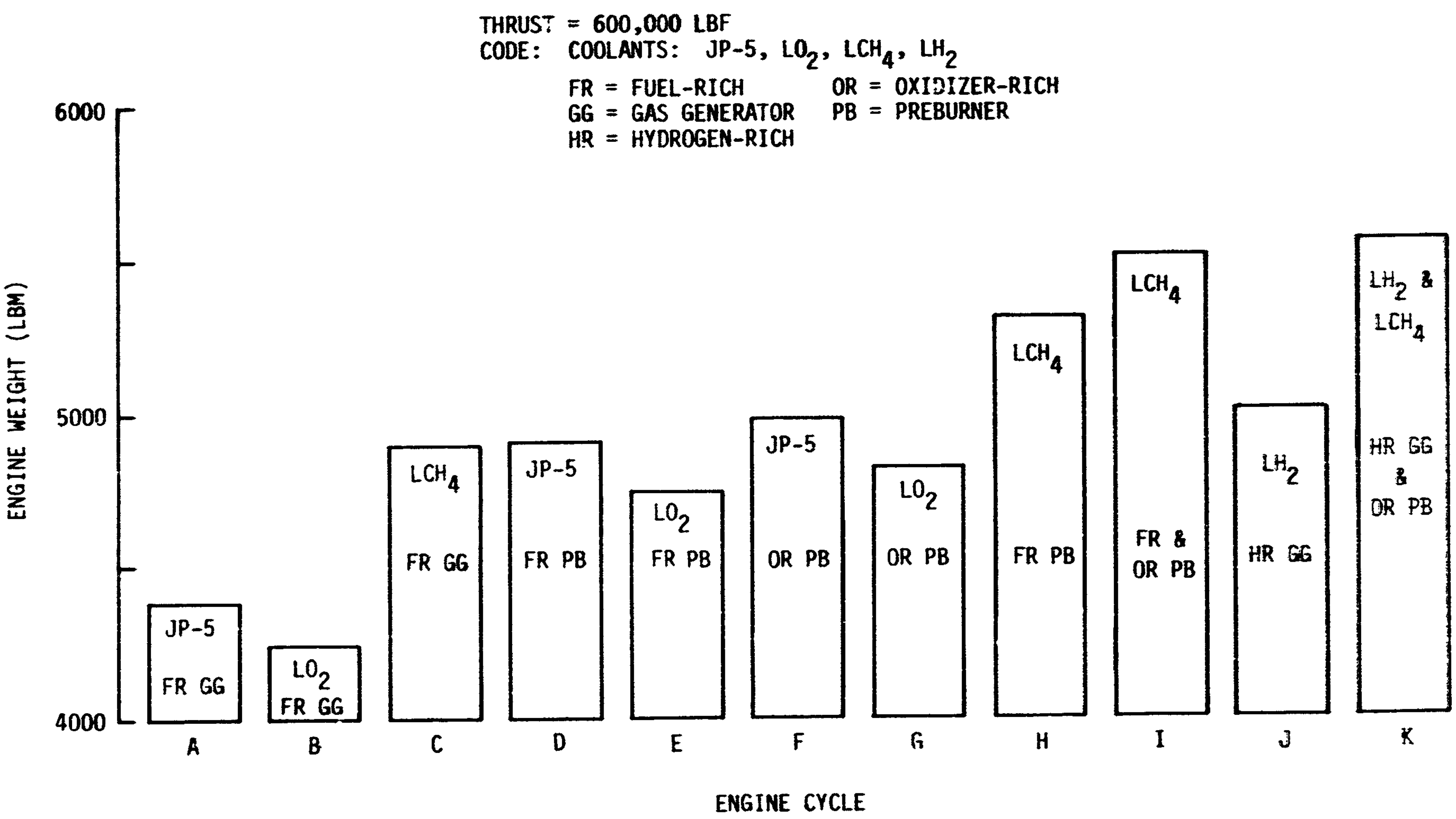



Figure 30. Weight Ranking of $LO_2$/HC Engine Cycles

The envelope ranking for the engine cycles is given in Figure 31. Despite the wide difference in chamber pressures (Pc - 2500 to 5000 psia), there is only a small difference in engine dimensions.

An engine cycle rating system was established based on the previously shown results and the Engine/Vehicle Trajectory Performance Assessment given in Section II.C. The rating system is shown in Table VII. The highest rating points were given to engine performance with a value of 5 points assigned to a specific impulse of 350 seconds. One point was assigned to an engine weight of 5000 pounds.

Chamber pressure was viewed as a life cycle influencing parameter, and, as such, lower pressures were awarded higher points. A nominal chamber pressure of 3000 psia was assigned 2 points. Hydrocarbon coking was envisioned as an unacceptable commodity. No coking, therefore, was assigned 3 points.

Interpropellant seals in turbomachinery require large amounts of inert gas, such as helium. Therefore, 3 points were assigned to a cycle without the requirement of an interpropellant seal. A shift in mixture ratio from the optimum value, such as required by gas generator cycles was penalized slightly, with the optimum mixture ratio assigned one point. Coolants such as $LO_2$ and $LCH_4$ were assigned 2 points and $LH_2$ coolant was assigned 3 points. The questionable coolant RP-1 was given one point.

Summation of the points assigned each cycle parameter leads to the rating given in Table VII. The dual throat cycle (J) because of its high performance, no interpropellant seal, and other features rates highest of all the cycles. Close competitors are cycles G through J.

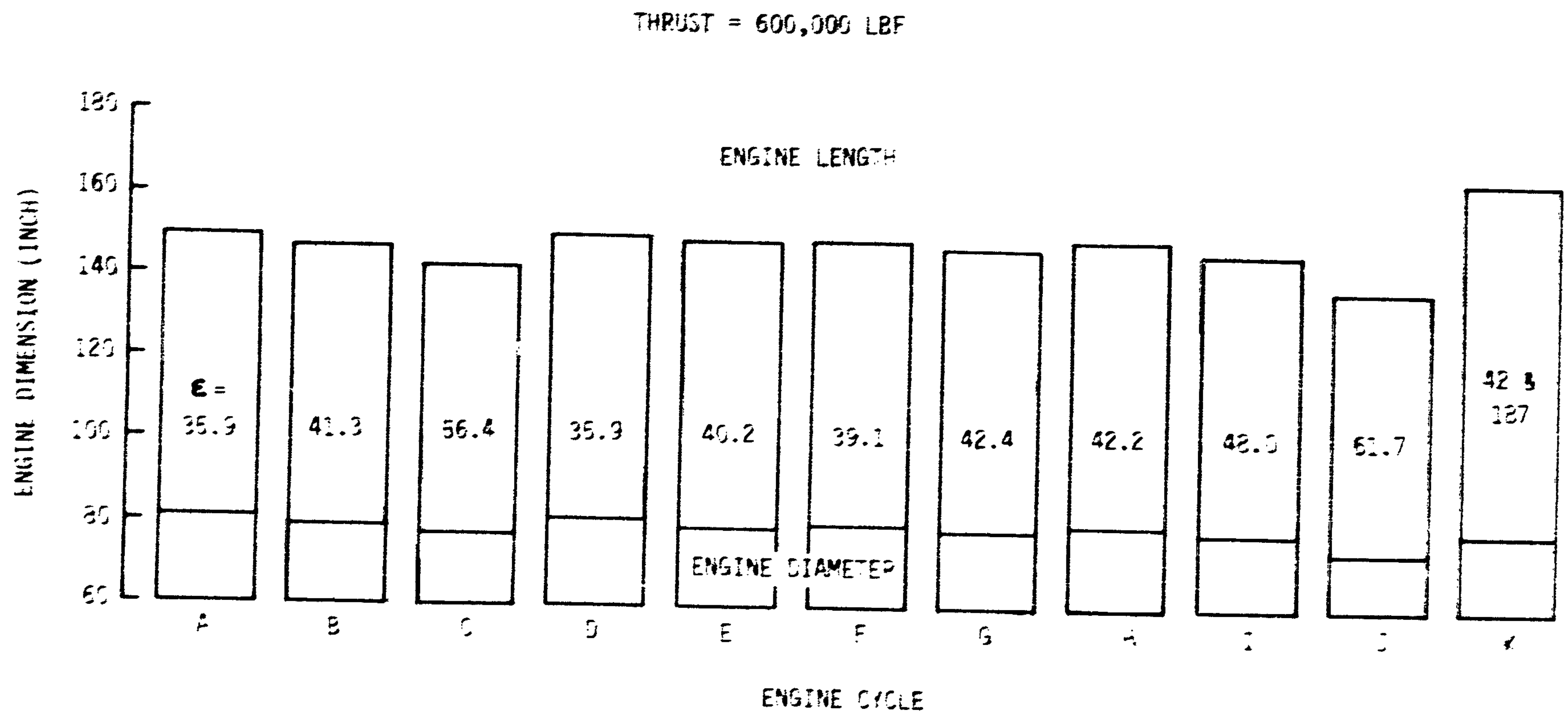



Figure 31. Envelope Ranking of $LO_2$/HC Engine Cycles

TABLE VII

$LO_2$/HC ENGINE CYCLE RATING

| Cycle | Fuel | Vacuum Performance Sec (Rating) | Engine Weight lbm (Rating) | Chamber Pressure psia (Rating) | Hydrocarbon Coking** | (Rating) | Inter-Propellant Seal(s) | Rating | Mixture Ratio (Rating) | Coolant Capability | (Rating) | Rating Maximum of 20 |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| A FC/FR/GG | RP-1* | 335 (-5) | 4380 (1.2) | 2500 (3) | Yes | (0) | Yes | (0) | 2.40 (.5) | Poor | (1) | 0.7 |
| B SC/FR/GG | RP-1 | 337 (-3.6) | 4250 (1.3) | 3000 (2) | Yes | (0) | Yes | (0) | 2.33 (.4) | Good | (2) | 2.1 |
| C FC/FR/GG | $LCH_4$ | 354 (4.4) | 4900 (1) | 4300 (-0.6) | ? | (2) | Yes | (0) | 2.88 (.2) | Good | (2) | 9.0 |
| D FC/FR/SC | RP-1* | 347 (3) | 4910 (1) | 2500 (3) | Yes | (0) | Yes | (0) | 2.80 (1) | Poor | (1) | 9.0 |
| E SC/FR/SC | RP-1 | 349 (4.4) | 4755 (1.1) | 2900 (2.2) | Yes | (0) | Yes | (0) | 2.80 (1) | Good | (2) | 10.7 |
| F FC/GR/SC | RP-1* | 348 (3.6) | 4960 (1) | 2800 (2.4) | No | (3) | Yes | (0) | 2.80 (1) | Poor | (1) | 12.0 |
| G SC/GR/SC | RP-1 | 350 (5) | 4830 (1.1) | 3100 (1.8) | No | (3) | Yes | (0) | 2.80 (1) | Good | (2) | 13.9 |
| H FC/FR/SC | $LCH_4$ | 358 (7) | 5315 (0.9) | 3000 (2) | ? | (2) | Yes | (0) | 3.50 (1) | Good | (2) | 14.9 |
| I FC/MR/SC | $LCH_4$ | 361 (9) | 5520 (0.8) | 3500 (1) | ? | (2) | No | (3) | 3.50 (1) | Good | (2) | 18.8 |
| J HC/HR/GG | RP-1 | 356 (9) | 5016 (1) | 5000 (-2) | No | (3) | Yes | (0) | [2.68] (.8) | Excel. | (3) | 14.8 |
| K MC/MR/MC | $LCH_4$ | 358 / 380 (11.2) | 5570 (0.8) | 2800 / 4000 (0) | ? | (2) | No | (3) | [3.35] (.8) | Good | (2) | 19.8 |
| REFERENCE | RP-1 | 350 (5) | 5000 (1) | 3000 (2) | No | (3) | No | (3) | 2.80 (1) | Good | (2) | 17 |

*JP-5 or purified RP-1 required
**High Turbine Temperatures may Alleviate Problem

II. Technical Discussion (cont.)

## B. TASK II - ENGINE PARAMETRIC ANALYSIS

Documentation of the results of this task was completed as scheduled. The results were previously reported in Bi-Monthly Progress Report 33452M-3.

## C. TASK III - ENGINE/VEHICLE TRAJECTORY PERFORMANCE ASSESSMENT

Based on recommendations from C. H. Eldred (NASA/Langley Research Center) the baseline two-stage vehicle proposed in Bi-Monthly Progress Report 33452M-2 was not utilized in the simplified trajectory analysis of this study. The two-stage vehicle utilized was taken from NASA/DOE Satellite Power Station studies ("Satellite Power System, Concept Development and Evaluation Program," Reference System Report DOE/ER-0023, October 1978). The characteristics of four NASA/Johnson Space Center configurations are given in Figure 32, compared with a Saturn V. The $LO_2$/RP-1 booster from Figure 32 delivering a 454 metric ton (1,000,900 lbm) payload was utilized in this study.

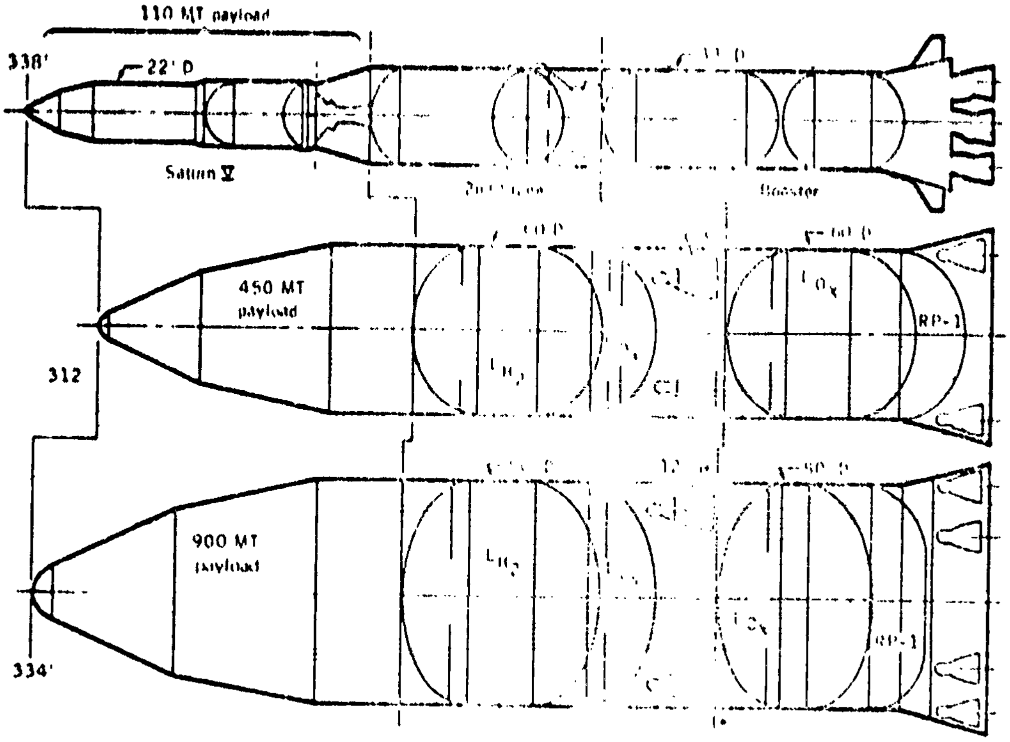
For $LO_2$/RP-1 engine cycles with thrust-to-weight ratios from 90 to 150, the following data apply:

$$\frac{\Delta W_{PL}}{\Delta (F/W_{ENG})} \quad 0.08 \text{ metric ton (176 lbm/lbf/lbm)}$$

$$\frac{\Delta W_{PL}}{\Delta I_S} \quad 1.5 \text{ metric ton/second (3,300 lbm/s)}$$

$$\frac{\Delta W_{ENG}}{\Delta I_S} \quad -7.7 \text{ metric ton/second (-17,000 lbm/s)}$$

These results indicate the importance of engine performance (specific impulse) for the first stage of the heavy lift vehicle and show the relative insensitivity of payload to engine weight for the two-stage vehicle.




| | $O_2$/RP-1 Booster | | $O_2$/Propane Booster | |
|---|---|---|---|---|
| Payload, tons, 90 x 500 km | 454 | 907 | 454 | 907 |
| Stage 1 inert, tons | 500 | 889 | 485 | 865 |
| Stage 1 propellant, tons | 4441 | 8236 | 4410 | 8177 |
| Stage 2 inert, tons | 233 | 400 | 245 | 421 |
| Stage 2 propellant, tons | 1937 | [illegible] | 2065 | 3832 |
| Gross lift-off weight, tons | 7565 | [illegible] | 7659 | 14203 |
| Number of engines, stage 1 | 12 | 24 | 12 | 24 |
| Number of engines, stage 2 | 6 | 12 | 6 | 12 |
| Staging altitude, km | 43.4 | 43.5 | 41.3 | 40.6 |
| Staging velocity (REL), km/sec | 1.84 | 1.91 | 1.70 | 1.78 |
| Booster maximum down range | 381 | 396 | 346 | 357 |

Figure 32. Two-Stage Ballistic Launch Vehicles

The $LO_2/LCH_4$ engine cycles show a similar data trend. However, the lower density methane causes a stage weight increase for the same gross liftoff weight. The stage weight increase nullifies most of the performance gain offered by the methane system. The final result is that the methane cycles offer only a slight payload increase (if any) despite the higher delivered performance.

D. TASK IV - BASELINE ENGINE SYSTEM DEFINITION

No activity scheduled.

## III. CURRENT PROBLEMS

Difficulty in getting the ALRC simplified trajectory program to compute the heavy lift vehicle two-stage trajectories has necessitated sole reliance on the R. Salkeld methodologies. The Salkeld methods have proven accurate to within one percent for single-stage-to-orbit trajectories, and are expected to provide representative results for the purposes of this study.

There will be a potential slip in the program schedule if approval for initiation of Task IV occurs after 27 June.

## IV. WORK PLANNED

A. TASK I

This task is completed.

B. TASK II

This task is completed.

IV. Work Planned (cont.)

C. TASK III

Complete documentation of this task.

D. TASK IV

Initiate this task after NASA approval.

E. TASK V

Conduct the Task I, II, and III review on 26 June at MSFC.